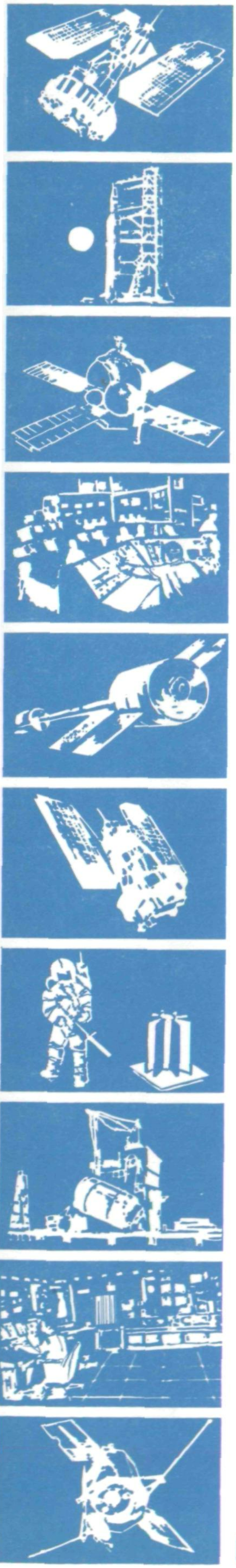
SPACE DIVISION



DIN: 73SD4220

N73 22725

NUCLEAR ELECTRIC PROPULSION

MISSION ENGINEERING STUDY

FINAL REPORT

CASE FILE COPY

VOLUME II

GENERAL ELECTRIC

March 1973

DOCUMENT NO. 73SD4220
MARCH, 1973

FINAL REPORT
NUCLEAR ELECTRIC PROPULSION MISSION ENGINEERING STUDY

COVERING THE PERIOD APRIL 1971 TO JANUARY 1973

VOLUME II - FINAL REPORT

PERFORMED UNDER
CONTRACT NO. JPL 953104

FOR

THERMIONIC REACTOR SYSTEMS PROJECT
PROPULSION RESEARCH AND ADVANCED CONCEPTS SECTION

JET PROPULSION LABORATORY
4800 OAK GROVE DRIVE
PASADENA, CALIFORNIA, 91103

This work was performed for the Jet Propulsion Laboratory, California Institute of Technology, sponsored by the National Aeronautics and Space Administration under Contract NAS7-100.

ENERGY SYSTEMS PROGRAMS

GENERAL ELECTRIC

SPACE SYSTEMS ORGANIZATION
Valley Forge Space Center
P. O. Box 8555 • Philadelphia, Penna. 19101

TABLE OF CONTENTS

TABLE OF CONTENTS (Cont)

TABLE OF CONTENTS (Cont)

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (CONT)

LIST OF TABLES

LIST OF TABLES (Cont)

ABSTRACT

This document summarizes the results of a mission engineering analysis of nuclear-thermionic electric propulsion spacecraft for unmanned interplanetary and geocentric missions. Critical technologies assessed are associated with the development of Nuclear Electric Propulsion (NEP), and the impact of its availability on future space programs. Specific areas of investigation include outer planet and comet rendezvous mission analysis, NEP Stage design for geocentric and interplanetary missions NEP system development cost and unit costs, and technology requirements for NEP Stage development. A multi-mission NEP Stage can be developed to perform both multiple geocentric and interplanetary missions. Development program costs for a 1983 launch would be of the order of $275 M, including hardware and reactor development, flight system hardware, and mission support. Recurring unit costs for flight NEP systems would be of the order of $ 25 M for a 120 kWe NEP Stage. Identified pacing NEP technology requirements are the development of 20,000 full power hour ion thrusters and thermionic reactor, and the development of related power conditioning. The resulting NEP Stage design provides both inherent reliability and high payload mass capability. High payload mass capability can be translated into both low payload cost and high payload reliability. NEP Stage and payload integration is compatible with the Space Shuttle.

SECTION 1
INTRODUCTION

The primary objective of the Nuclear Electric Propulsion (NEP) Mission Engineering Study is to perform a mission engineering study of nuclear-thermionic electrically propelled spacecraft for unmanned interplanetary and geocentric missions to determine the implications of Nuclear Electric Propulsion on future space programs. This volume of the study final report presents the NEP Stage, design status, mission operations, and costs. The stage is designed to perform both interplanetary science missions and geocentric orbit missions, involving the transportation of operational payloads, such as communication satellites, to and from geocentric earth orbit. The NEP Stage configuration and the mission profiles and operations are presented, based on defined mission objectives. Gross Ground Support Equipment (GSE) and operational equipment are identified.

1.1 PURPOSE AND SCOPE

This effort is directed toward the definition of a Nuclear Electric Propulsion (NEP) Stage for interplanetary and earth orbital missions. The NEP stage consists of a propulsion system, plus the onboard guidance and control, communications, and data storage and logic, necessary to provide an autonomous stage with full multi-mission capability. The mission operations required for both interplanetary and geocentric earth orbit missions are identified and mission performance evaluated. Necessary ground support equipment, operational equipment, and support facilities are defined.

1.2 KEY GUIDELINES AND CONSTRAINTS

The key guidelines and constraints used in the assessment of nuclear electric propulsion for interplanetary and geocentric earth orbit missions are shown in Table 1-1. Emphasis is placed on multi-mission capability from a spacecraft design based on current or near term technology to maximize cost effectiveness and minimize propulsion system development costs.

Table 1-1. 120 kWe NEP Stage Mission Engineering Study Program Guidelines and Constraints

<u>Interplanetary Missions</u>
Shuttle-Centaur D-IT Baseline Launch Vehicle
Maximum Use of Previous Trajectory Analysis
High Thrust (Chemical) Injection to Earth Escape
Low Thrust Terminal Propulsion
Comet Halley Rendezvous and Multiple Outer Planet Exploration
<u>Geocentric Orbit Missions</u>
Synchronous Equatorial Earth Orbit Baseline Mission
Shuttle/Shuttle-Chemical Tug Baseline Launch Vehicles
<u>Both Missions</u>
Maximum Use of Previous Propulsion System Design Studies
Employ Realistic Level of Technology
Define Mission Operations
Define GSE and Support Facilities
Emphasis on Impact of Nuclear Electric Propulsion on Mission Operations

Specific guidelines and constraints for the design of the multi-mission NEP Stage may be found in Volume II, Appendix A - Design Specification for the Thermionic Nuclear Electric Propulsion Multi-Mission Stage.

The key guidelines and constraints utilized in the preliminary design of the avionics subsystem are shown in Table 1-2. A geosynchronous earth orbit mission is assumed. Many of the components of the avionics subsystem are directly applicable to interplanetary missions. Differences will be in the selection of attitude control sensors, implementation of data handling hardware, software for the Thrust Vector Control (TVC) steering maneuvers, communication requirements, and certain components for functions peculiar to the geosynchronous orbit mission.

Table 1-2. Avionics Subsystem Design Guidelines/Constraints

Design
Long Life: -50,000 Hours in Space Environment
Commonality with Interplanetary Missions
Maximum Utilization of Electric Propulsion for Attitude Control
Meet Shuttle Cargo Bay Geometrical Envelope
Accommodate Varying Degree of Docking Target Cooperation

An operating lifetime for the general avionics subsystem functional subsystems has been established at 50,000 hours. The video/illumination and scanning laser radar subsystem and the video transmitter are required to be operational only during rendezvous and docking maneuvers and consequently will be operating at maximum capacity for only a short period of time. The 50,000 hour life requirement is considered to be easily achieveable with the type of electronics and other active components being considered. In fact, lifetime specifications for most projected geosynchronous communication satellites programs are in the range of 7 to 10 years. Refueling for both the primary and auxiliary propulsion systems will be accomplished at the end of each round trip by docking with the Propellant Logistics Depot (PLD). This approach maximizes the NEP Stage payload capability, since the propellant carried for each orbit transfer mission is minimized.

An auxiliary thrust vector control system, and auxiliary thruster are required to perform docking maneuvers for the geocentric missions. This system is integrated with the ion engine system. The combined system accommodates all TVC and attitude control functions.

The overall avionics subsystem configuration dimensions are compatible with the Shuttle cargo bay dimensions, 4.6 m diameter. The basic mission is to deliver new payloads and retrieve malfunctioning or spent payloads. The critical avionics subsystem desigh function is to accommodate rendezvous and docking with potentially uncooperative payloads in

geosynchronous orbit. Three degrees of cooperation are considered, defining a baseline cooperative system, and adding components and increasing functional complexity to accommodate increased docking complexity.

1.3 NEP STAGE SUMMARY

Assessment of currently established mission performance and versatility requirements resulted in the definition of an end thrust NEP Stage design. The general arrangement and key system parameters of the 120 kWe, end thrust NEP Stage are presented in Figure 1-1. Electrical power is provided by a 23-volt internally fueled thermionic reactor. To provide 120 kWe to the thrust subsystem, the reactor generates approximately 1580 kW of thermal power, converting approximately 136 kWe to electrical power, and rejecting the rest as waste heat via a pumped primary coolant loop and a heat pipe primary radiator.

The end thrust NEP Stage is basically a conical configuration with a cylindrical primary radiator. The reactor is boomed to minimize shielding and ion engine interactions. An array of 30 cm mercury electron bombardment ion engines provides axial thrust at a variable specific impulse of 4000 to 5000 sec.* The thruster array is composed of 24 engines, including 20 percent spares, canted at nine degrees to reduce ion engine exhaust interactions.

*For geocentric orbit applications, the NEP Stage operates at a specific impulse of 4000 sec. The specific impulse is increased to 5000 sec for interplanetary missions. The use of 3000 seconds specific impulse, although desirable for geocentric missions because trip times are reduced, is precluded by the large ion engine array areas, and related shielding and shuttle packaging problems.

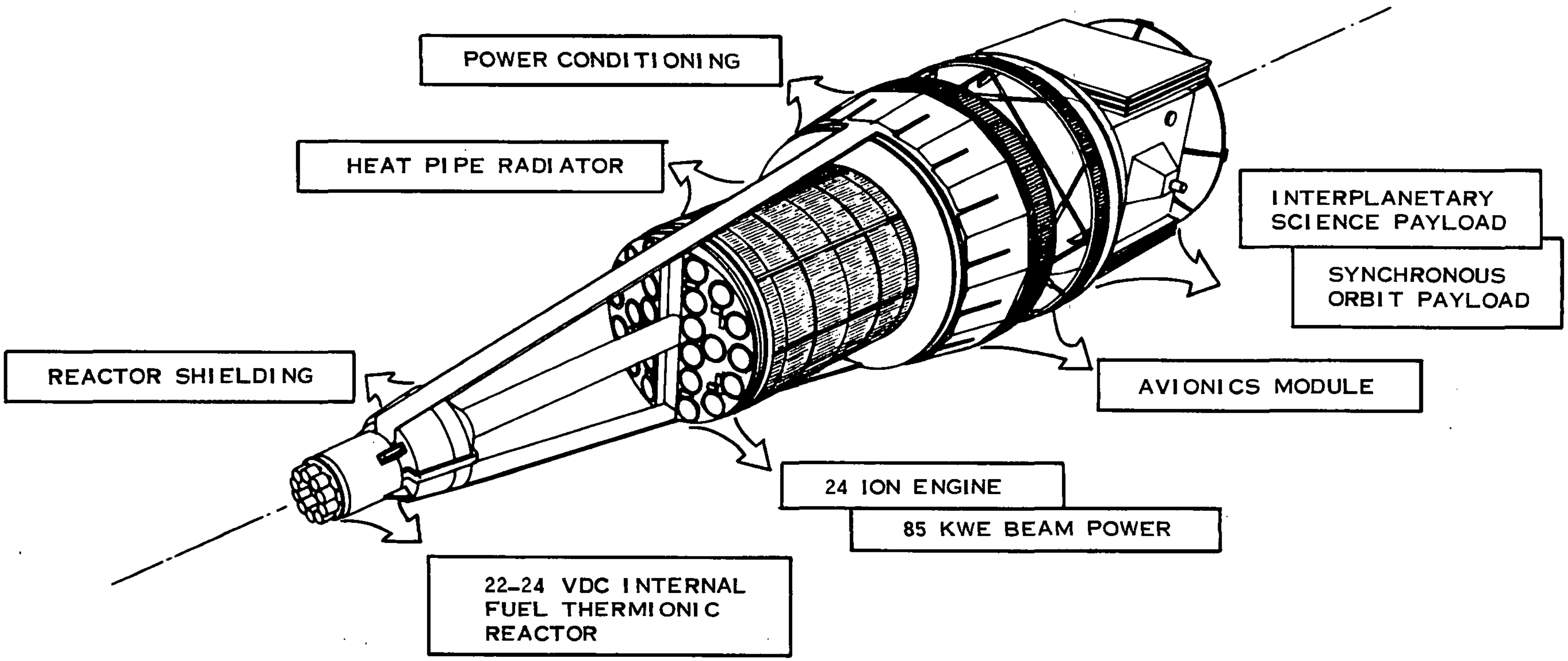


Figure 1-1. Multi-Mission NEP Thermionic Stage (120 kWe)

The basic NEP Stage including the avionics package is approximately 12.8 m long at a maximum diameter of 4.6 m. The specific mass of the NEP Stage is 32 kg/kWe based on the net power delivered to the thrust subsystem.

The major NEP Stage subsystem masses are:

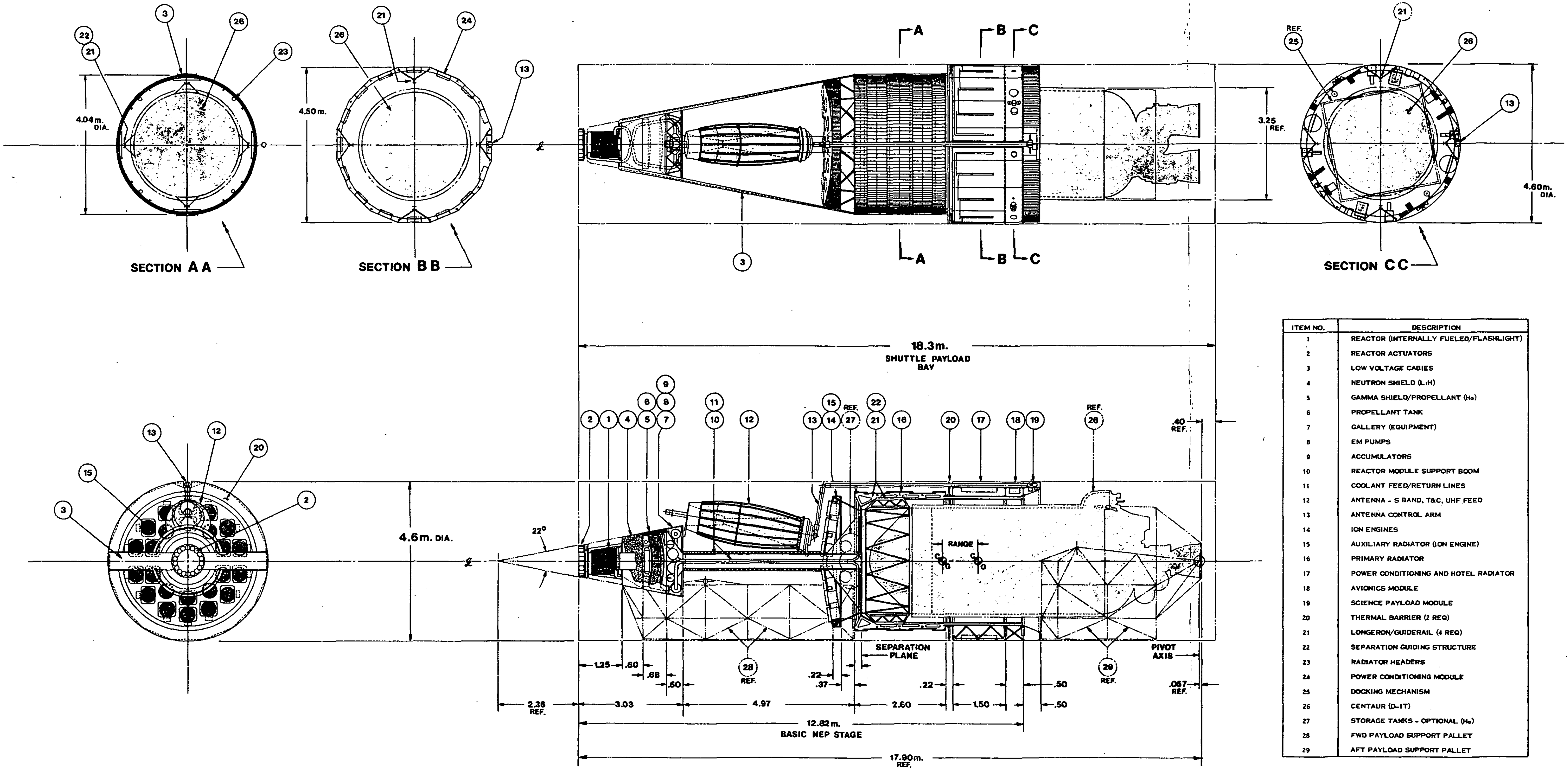
1.	Power S/S	3030 kg
2.	Thrust S/S	755 kg
3.	Propellant S/S	5740 kg *
4.	Avionics S/S	460 kg

The defined NEP Stage configuration represents a 1983 IOC. A 30,000 full power hour growth version of this stage could be available for a 1986 IOC. The growth version would utilize one flashlight thermionic reactor that delivers $\leq$ 240 kWe to the thrust subsystem. In addition, projected technology advances would permit the allowable mercury ion engine beam current to be doubled, resulting in about the same ion engine array area.

The conceptual design of the 120 kWe NEP Stage, integrated with a Centaur kick stage, is presented in Figure 1-2. This is the launch configuration for interplanetary missions. As noted in Figure 1-3, the center-of-gravity location of the NEP Stage/Centaur interplanetary configuration with mercury stored in the aft tanks is compatible with the Shuttle launch requirements. Figure 1-4 shows the NEP Stage packaged in the Shuttle cargo bay for geocentric orbit missions. The figure indicates that NEP Stage/geocentric payload configurations studied are compatible with the Space Shuttle payload center-of-gravity envelope. Other geocentric mission payloads would have to be assessed individually for Shuttle launch feasibility. Figure 1-5 presents the Dual Mode NEP Stage launch configuration packaged in the Shuttle cargo bay.

---

* For interplanetary missions



ITEM NO.	DESCRIPTION
1	REACTOR (INTERNALLY FUELED/FLASHLIGHT)
2	REACTOR ACTUATORS
3	LOW VOLTAGE CABIES
4	NEUTRON SHIELD (LiH)
5	GAMMA SHIELD/PROPELLANT (Hg)
6	PROPELLANT TANK
7	GALLERY (EQUIPMENT)
8	EM PUMPS
9	ACCUMULATORS
10	REACTOR MODULE SUPPORT BOOM
11	COOLANT FEED/RETURN LINES
12	ANTENNA - S BAND, T&C, UHF FEED
13	ANTENNA CONTROL ARM
14	ION ENGINES
15	AUXILIARY RADIATOR (ION ENGINE)
16	PRIMARY RADIATOR
17	POWER CONDITIONING AND HOTEL RADIATOR
18	AVIONICS MODULE
19	SCIENCE PAYLOAD MODULE
20	THERMAL BARRIER (2 REQ)
21	LONGERON/GUIDERAIL (4 REQ)
22	SEPARATION GUIDING STRUCTURE
23	RADIATOR HEADERS
24	POWER CONDITIONING MODULE
25	DOCKING MECHANISM
26	CENTAUR (D-1T)
27	STORAGE TANKS - OPTIONAL (Hg)
28	FWD PAYLOAD SUPPORT PALLET
29	AFT PAYLOAD SUPPORT PALLET

Figure 1-2. 120 kWe NEP Stage Interplanetary Mission Launch Configuration

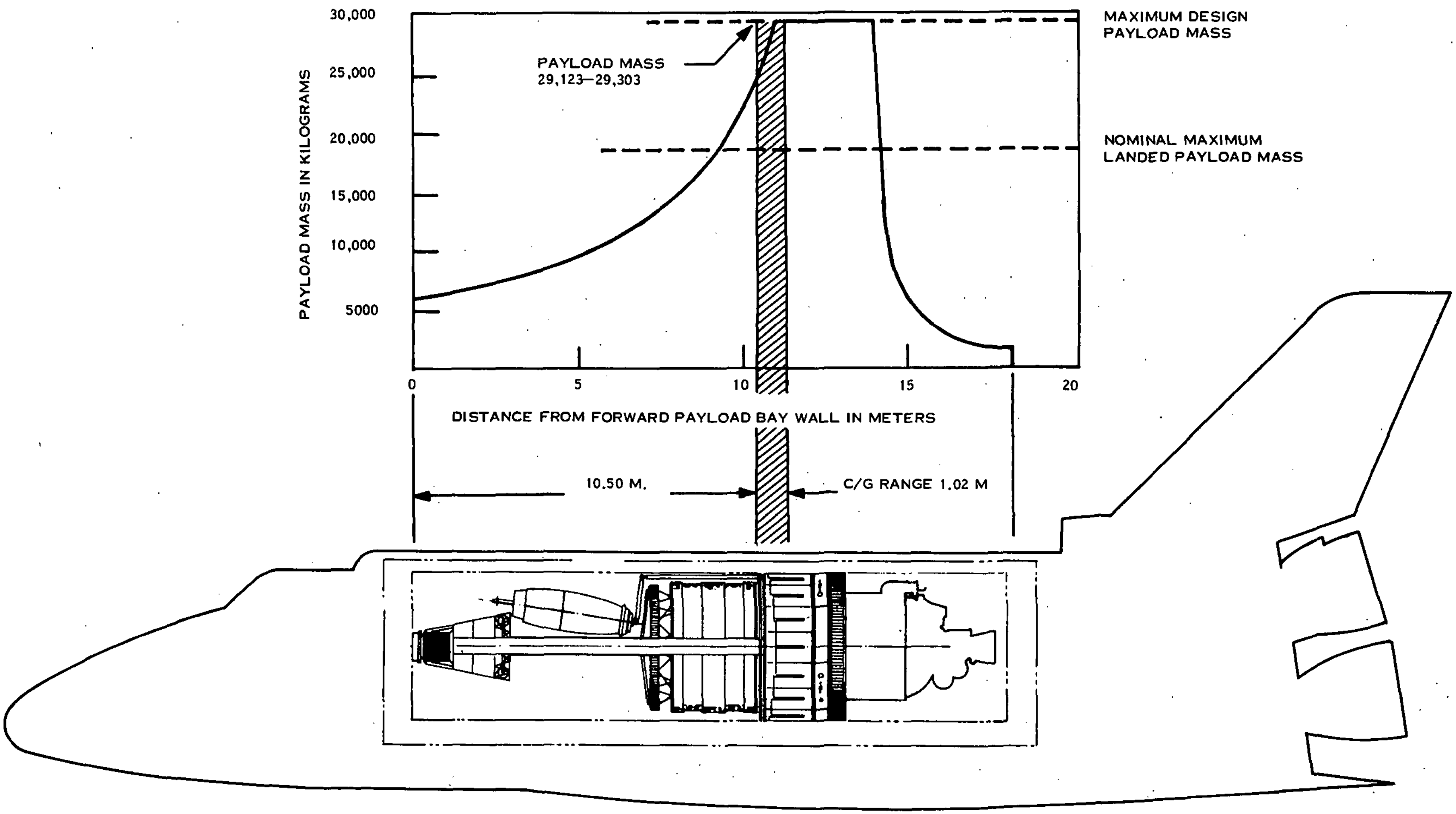


Figure 1-3. 120 kWe NEP Stage Interplanetary Mission Launch Configuration Center-of-Gravity Location

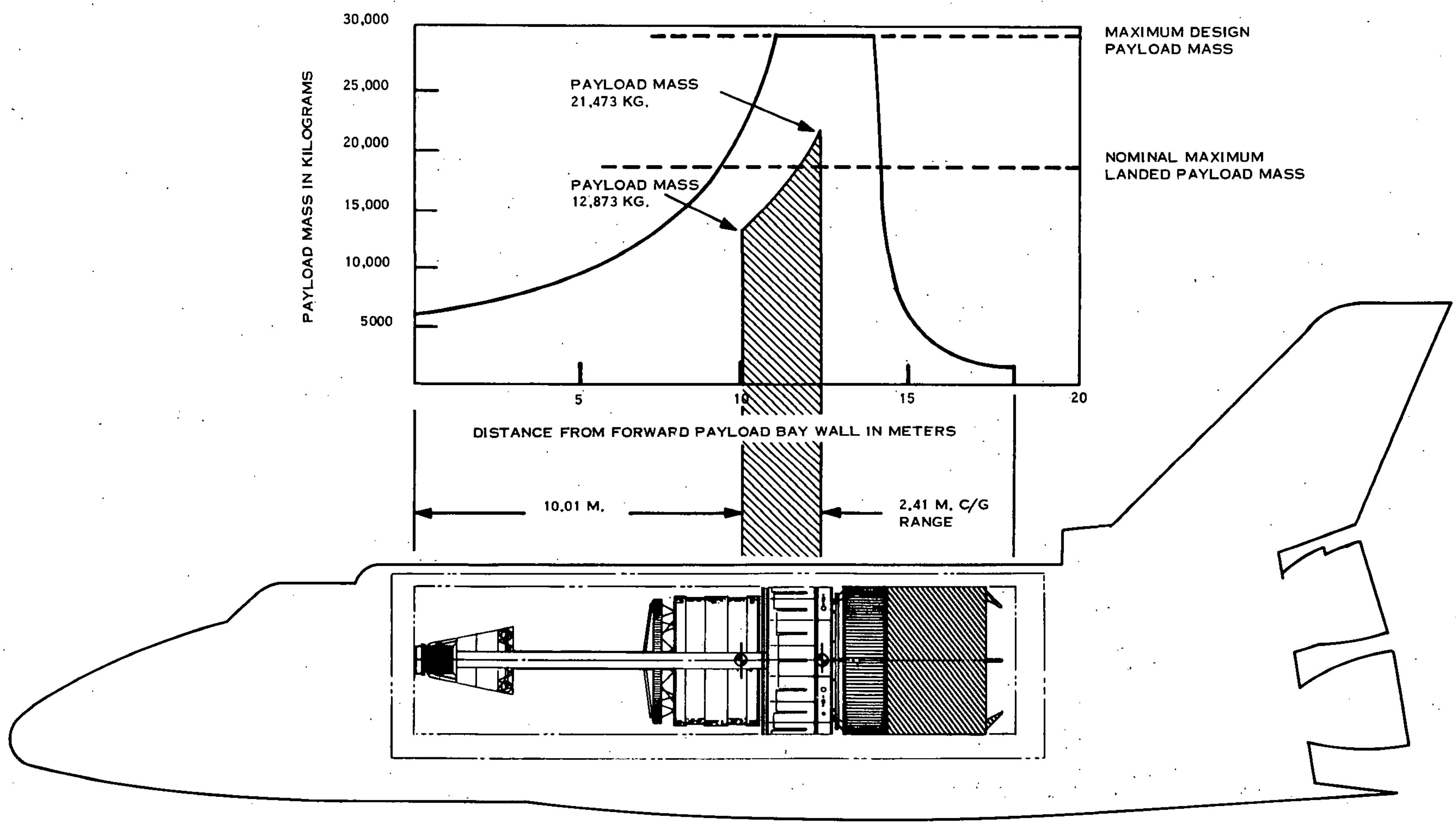


Figure 1-4. 120 kWe NEP Stage Geocentric Mission Launch Configuration

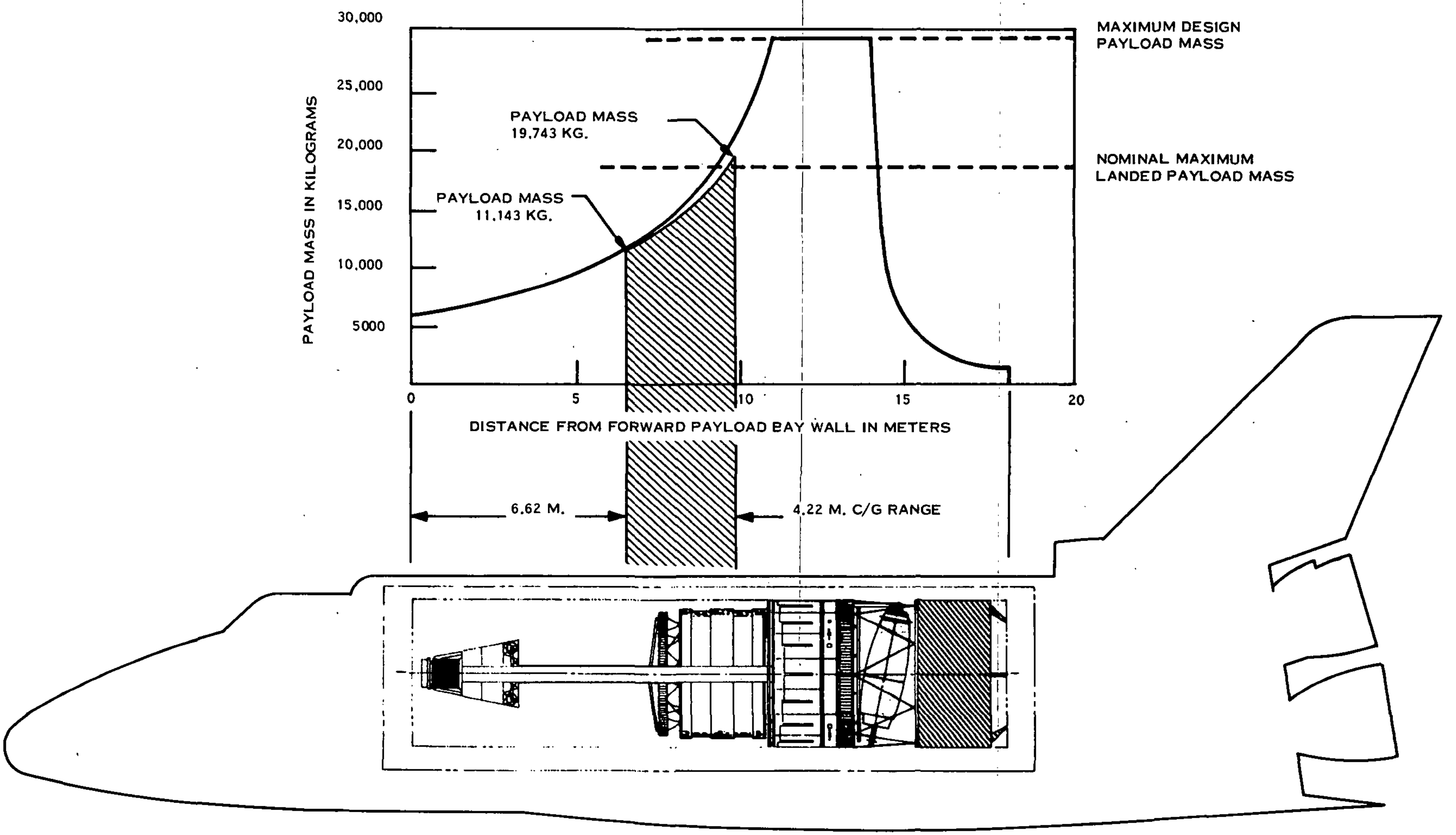


Figure 1-5. 120 kWe NEP Stage Dual Mode Shutter Integration

The characteristics of the avionics subsystem are summarized in Tables 1-3 and 1-4.

Table 1-3. Avionics Subsystem Summary Characteristics

Disk-Shaped Configuration (4.6 m x 0.5 m)
3-Axis Attitude Control
Earth Orbit Normal Reference
Ground Control Data Processing Capability
Hydrazine Reaction Control Subsystem
Autonomous Scanning Laser Radar (SLR) for Rendezvous/Docking
Video System for Docking
Communications at S-Band-Omnidirectional Capability
Avionics Subsystem Mass - 460 kg

Table 1-4. Avionics Subsystem Performance

	Performance Data		
Subsystem	Size (m^3)	Mass (kg)	Power (w)
Attitude Control	0.03	34	83
Auxiliary Propulsion	0.1	98	35
Communications	0.05	62	322
Video/Lighting Platform	0.1	15	20
Scanning Laser Radar	-	16	155
Structure	NA	160	NA
Thermal	NA	25	-
Mechanisms	0.1	20	-
Power Distribution	0.03	30	NA
Total	0.31	460	615

1.4 MISSION SUMMARY

Based on interplanetary mission analysis, the Comet Halley rendezvous and a Jupiter orbiter mission were selected as the baseline NEP interplanetary missions. The Comet Halley rendezvous mission (Figure 1-6) with a trip time of 900 days requires a low thrust propulsion time of 18,000 hours and an initial hyperbolic excess velocity of 2.5 km/sec. For this mission, the NEP Stage is launched to earth escape in January-June 1983, with comet rendezvous in December 1985, 50 days before perihelion. This marks the beginning of approximately 100 days of scientific observation within the environs of the comet.

The Comet Halley mission is characterized by an accelerate-decelerate-accelerate electric propulsion thrust profile. The comet orbit is retrogate and approximately 18 degrees (Reference 1-1a) out of the ecliptic. (This feature is exaggerated in Figure 1-6).

The baseline Jupiter orbiter mission, depicted in Figure 1-7, requires 14,000 hours of propulsion time, corresponding to a trip time of 900 days. The Centaur D-1T provides a hyperbolic excess velocity of 2.9 km/sec during earth escape. Of the 900-day trip, 158 days are utilized to effect descent to a circular orbit of 5.9 Jupiter radii. Since the NEP Stage descends in a slow, nearly circular spiral trajectory, scientific observations can be made throughout the descent, as well as from the terminal orbit. Alternately, the inward spiral could be temporarily terminated, as appropriate, to permit detailed examination of the Jovian moons.

The example baseline NEP Stage mission selected for geocentric orbit applications is the transportation of operational payloads to and from synchronous equatorial earth orbit. The mission profile for this application is shown in Figure 1-8. The NEP Stage is Shuttle launched to low earth orbit with a Propellant Logistics Depot (PLD) which stores enough mercury propellant, hydrazine for the attitude control subsystem, and other consumables for the 20,000 hour NEP Stage operational lifetime. The NEP Stage with PLD attached spirals out to a 14,800 by 35,800 km intermediate parking orbit (15 degree orbital inclination) from where it can conduct approximately ten round trip missions to geosynchronous orbit. The Shuttle/Chemical Tug conducts round trip flights to the intermediate orbit

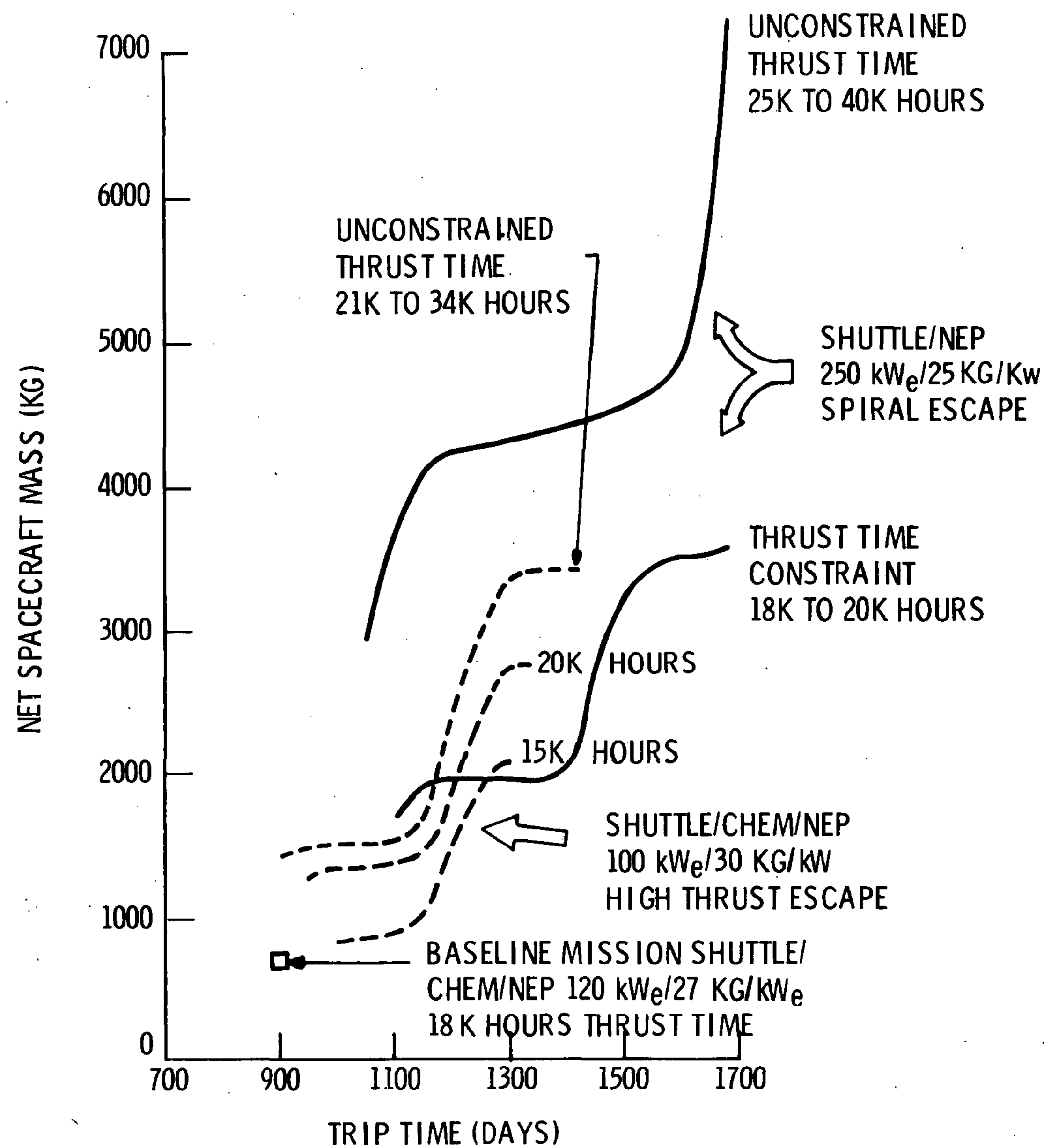


Figure 1-6. Comet Halley Rendezvous Mission (900 Days; 18,000 Full Power Hours)

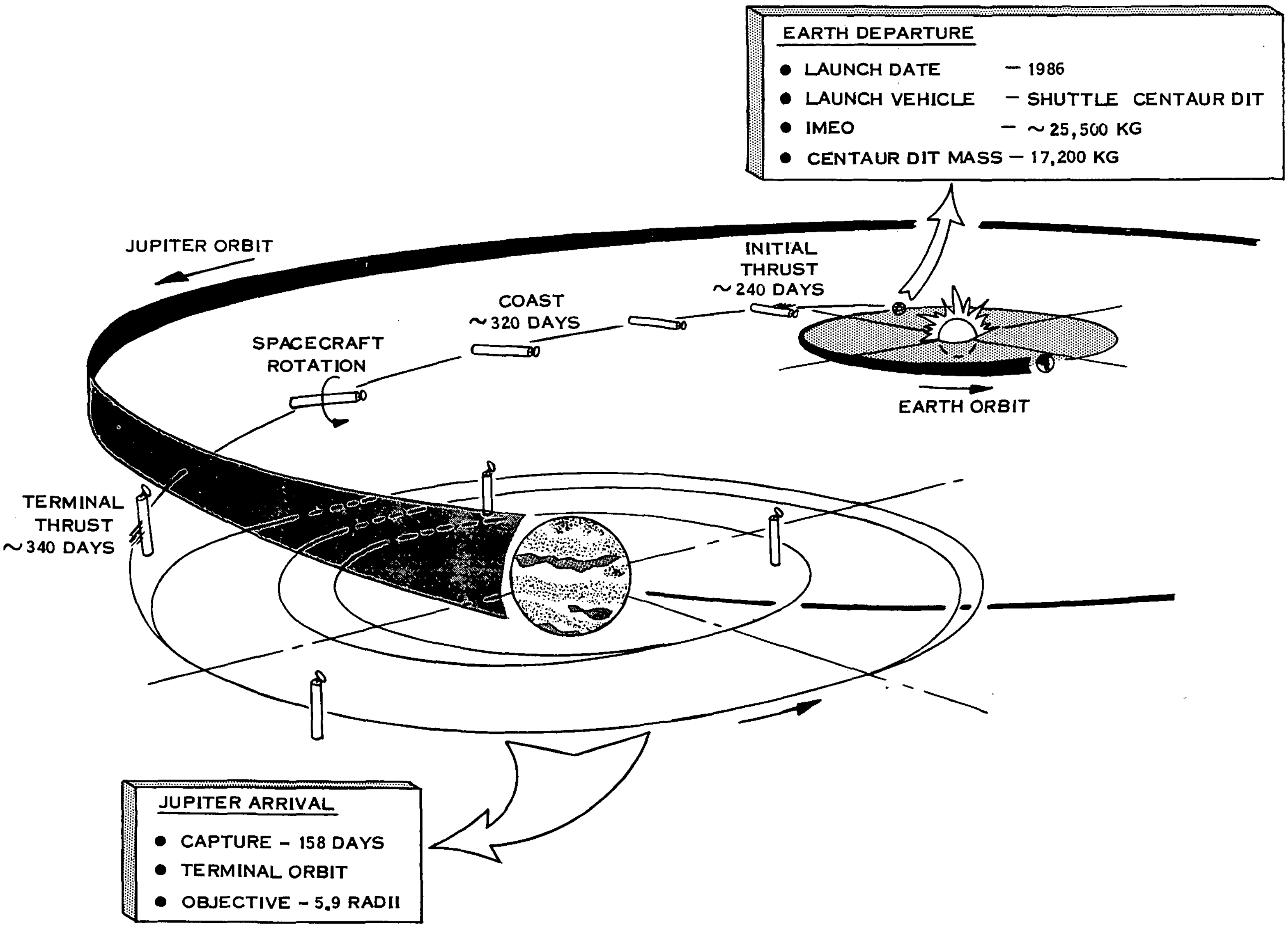


Figure 1-7. Jupiter Orbiter Mission (900 Days; 14,000 Full Power Hours)

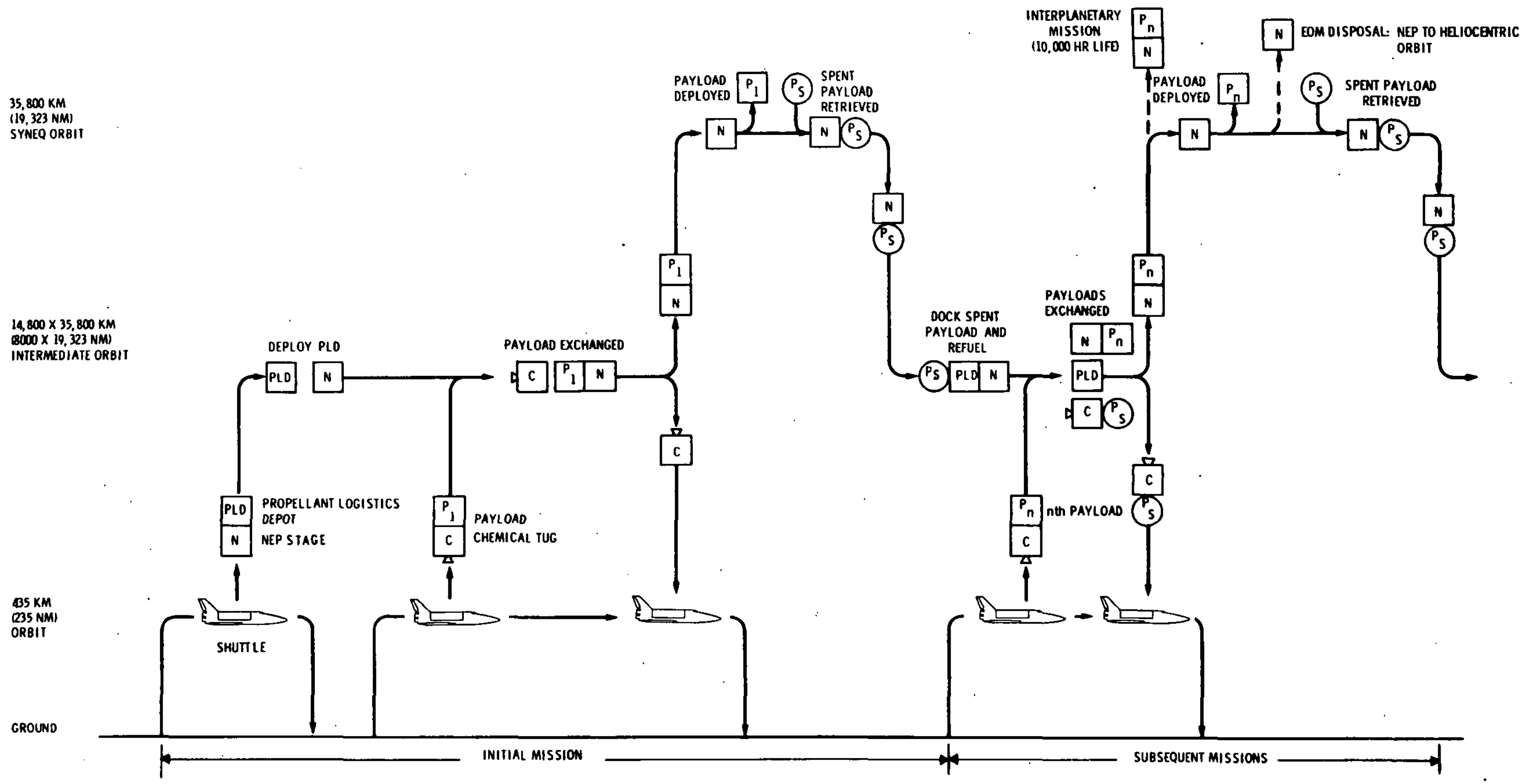


Figure 1-8. Baseline NEP Stage Geosynchronous Orbit Mission Profile

to deliver new synchronous orbit payloads to the NEP Stage and to return spent payloads to earth for possible refurbishment.

The 14,800 by 35,800 km intermediate parking orbit is selected because it is above the Van Allen radiation belt and permits a direct comparison of NEP Stage performance with current Solar Electric Propulsion (SEP) geocentric mission studies. * This mission profile minimizes the exposure of the synchronous orbit payload to Van Allen radiation (because of the minimum transfer time obtainable with the Chemical Tug), reduces the trip time to synchronous orbit (relative to an all NEP mission mode), and increases the payload capability to synchronous orbit (relative to that obtainable with the Chemical Tug alone).

After the NEP Stage has completed its 20,000 full power hour life, it can be used to insert itself into a heliocentric orbit for safe disposal. The option also exists for the NEP Stage to perform an interplanetary mission after completing up to 10,000 full power hours in geocentric orbit.

The most significant conclusion obtained from the mission analysis effort is the practicability of an interplanetary multimission NEP Stage. This spacecraft is capable of performing not only both baseline interplanetary missions, the Comet Halley rendezvous and the tight Jupiter orbiter, but a large family of outer planet exploration missions as well.

For interplanetary missions, the Shuttle/Centaur D-1T launch vehicle provides improved mission performance relative to the Titan/Centaur family, except for the Titan III L4/Centaur. For outer planet missions, trip time and propulsions time are not oversensitive to increases in NEP Stage specific mass (a 5 kg/kWe specific mass increase results in approximately a ten percent increase in trip time and propulsion time), which may be expected to occur during the NEP Stage development program. Such increases must be minimized.

*APC Committee Study - Phase I, 1972.

Specific impulse requirements for interplanetary mission applications do not exceed 5000 seconds, which should simplify the development of the main power conditioning since the output voltage will be no greater than about 3000 Vdc.

Based on the reference geocentric orbit mission profile, the NEP Stage has the capability to deliver (and return) up to ~ 8600 kg between a 14, 800 x 35, 800 km intermediate orbit (15 degree inclination) and synchronous equatorial orbit in about 100 days round trip flight time. Over the 20, 000 full power hour lifetime, the total payload capability is 58, 000 kg, including allowance for the time to spiral out from the Shuttle deployment orbit.

Power levels above 120 kWe require further evaluation for other mission modes, such as all-electric propulsion geocentric NEP Stage missions.

For interplanetary application, the mission affects the NEP operational procedures only during the final stages of the flight, when navigational and course correction procedures will depend on the type of target. Comet intercept accuracy requirements are considerably more stringent than planet flyby or orbiting missions, so navigation and trajectory corrections will have to be correspondingly more frequent and precise. If the vehicle is to fully investigate the comet, the relative position of a comet's nucleus and tail sections, in relation to the sun's position the path approach velocity of the NEP Stage relative to the comet must be known.

A number of NEP Stage design details, operational conditions and equipment need further identification in order to delineate mission operations in greater detail. These include:

1. Launch windows
2. Required tests and checkouts during fabrication and prelaunch
3. Integration of NEP Stage and Centaur (kick stage)
4. Need for NEP Stage preheat and effect on launch pad safety
5. Need for auxiliary power source in shuttle cargo bay

1.5 GROUND SUPPORT EQUIPMENT SUMMARY

Table 1-5 lists the identified Ground Support Equipment (GSE) and Operational Equipment required to support NEP Stage operations. All GSE Operation Equipment identified are required, whether the mission is interplanetary or geocentric, except for the Centaur support equipment, a Chemical Tug/Synchronous Orbit Payload Transfer Module, and the Propellant Logistics Depot (PLD). The latter two items of Operational Equipment, however, are dependent upon the geocentric orbit mission profile selected and are not required for other identified NEP geocentric orbit mission modes.

It appears that one (or at the most two) single avionics subsystem can be developed that will be used to perform all identified missions. Only minor variations in the science between comet rendezvous missions and planetary missions have been identified. For geocentric orbit missions, many of the components of the avionics subsystem will be directly applicable to interplanetary missions. Major differences will be in the selection of attitude control sensors, implementation of data handling hardware, software for the Thrust Vector Control (TVC) steering maneuvers, communication requirements, and certain components for function peculiar to the geocentric orbit mission.

The only unique hardware development that may be required for the NEP geocentric orbit operations is that involved with in-orbit refueling of the NEP Stage.

Any payload to be transported by the Space Shuttle is subject to the normal operational Shuttle-induced environments, in addition to various potential accident environments. During normal operation, the environment within the Shuttle cargo bay is relatively mild compared to that of other unmanned launch vehicles.

Safety and handling can be improved, and support requirements imposed on the Shuttle reduced, if a transfer module is used to support the NEP Stage within the cargo bay of the Shuttle. The transfer module is a carriage-type assembly in which the NEP Stage is placed before being installed in the Shuttle. The entire Stage/transfer module assembly is placed in the Shuttle cargo bay. By using such an assembly, the integration items

Table 1-5. Ground Support Equipment and Operational Equipment Requirements

Ground Support Equipment

- Fabrication and Test
  - TFE Test Equipment
  - Leak Test and Weld Inspection Equipment
  - NaK Charging and Purification Facility
  - Hot Test Facilities
  - Avionics Subsystem Simulator(s)
  - Low Voltage Electric Power Source
  - High Voltage Electric Power Source
  - Test Facility for Ion Engine Array Performance Test
  - Ion Engine Electrical Load Simulator
  - Propulsion System Simulator for Avionics Subsystem Test
  - Handling Rigs and Transporters for each Subsystem
  - Shipping Storage Containers with Environmental Control Package for each Subsystem
  - Shipping Container for Assembled NEP Stage
- Arrival at Launch Site and Prelaunch
  - Nuclear Storage and Checkout Facility
  - Checkout Equipment for NEP Systems
  - Alkali Metal Handling Facility
  - Mercury Propellant Handling Facility
  - Handling Equipment
  - Transporter
  - Inert Gas Supply and Handling Facilities
- Launch-Mission Completion
  - Space Flight Operations Facility

Operational Equipment

- NEP Stage Transfer Module
- Chemical Tug-Synchronous P/L Transfer Module
- Propellant Logistics Depot

required for the space transportation of the NEP Stage, such as thermal control and electrical power, can be incorporated into the transfer module rather than being designed into the Stage or the Space Shuttle.

A considerable amount of test equipment will be required in conjunction with propulsion system fabrication. It is recommended that a reactor power system storage and checkout facility be available at the launch site.

SECTION 2
NEP STAGE CHARACTERISTICS

Previous studies (References 2-1 and 2-2) have provided the preliminary design definition of a Nuclear Electric Propulsion (NEP) system to perform unmanned comet rendezvous and outer planet exploration missions. The Geocentric Orbit mission represents application for nuclear electric propulsion systems which had not been previously evaluated. The in-core thermionic reactor power system is the leading nuclear power system candidate for these electric propulsion applications. Thermionic power systems are similar to solar power systems in that they consist of many static power conversion modules arranged to tolerate module failure. A heat rejection system providing a high degree of redundancy can be incorporated with acceptable weight penalties. The thrust system is also modularized to tolerate failures by providing a separate power conditioning system for each ion engine with an assumed 20 percent redundancy in these modules. This potential for high reliability, in addition to low specific weights, makes the thermionic reactor electric propulsion system attractive for both interplanetary and geocentric orbit applications.

The NEP Stage description and characteristics are presented in this section. The key configuration drivers are discussed, and the results of the NEP Stage configuration analysis are presented. Alternate NEP Stage configurations are identified.

2.1 NEP STAGE DEFINITION

The multi-mission Nuclear Electric Propulsion Stage defined for interplanetary and geocentric orbit mission applications consists of a power subsystem, thrust subsystem, propellant subsystem, and an avionics subsystem. The major subsystems and components that are included in these systems are indicated in Figure 2-1. The power subsystem and thrust subsystem comprise the propulsion system. The subsystems and components that make up the NEP Stage power, thrust, and propellant subsystems are common for both interplanetary and geocentric orbit missions. The subsystems that comprise the avionics subsystem have a high degree of commonality for both types of applications. However, the docking requirement is unique to geocentric orbit missions.

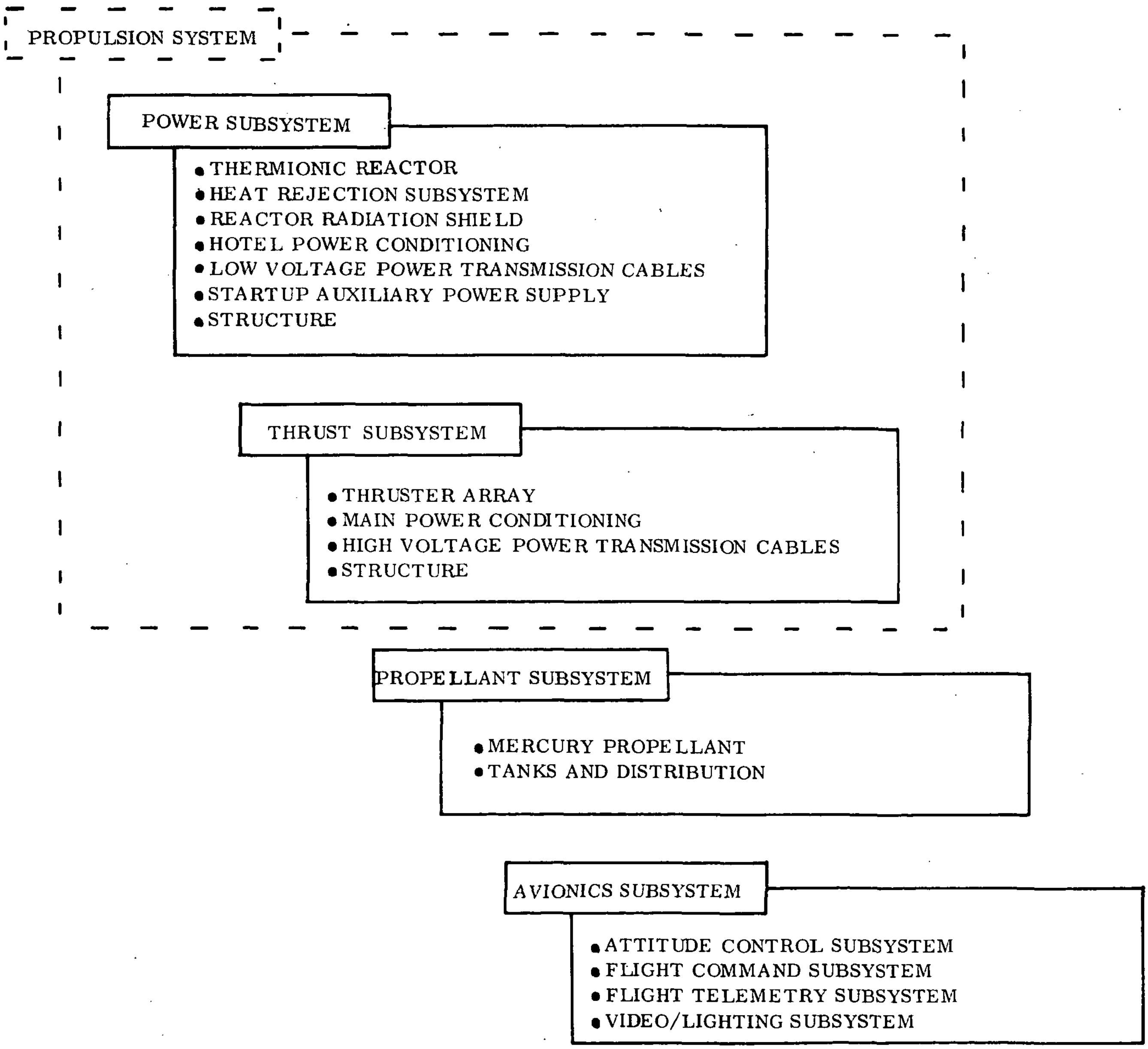


Figure 2-1. NEP Stage Definition

The specific mass of the NEP Stage is based on the propulsion system. The mass of the mercury propellant and tankage and the avionics system does not contribute to the overall stage specific mass.

2.2 KEY CONFIGURATION DRIVERS

In arriving at a preliminary conceptual design for the NEP Stage, it became apparent that several NEP system and mission related interfaces were going to have a profound impact on the final configuration. These key configuration drivers are listed in Figure 2-2, and are discussed in following subsections.

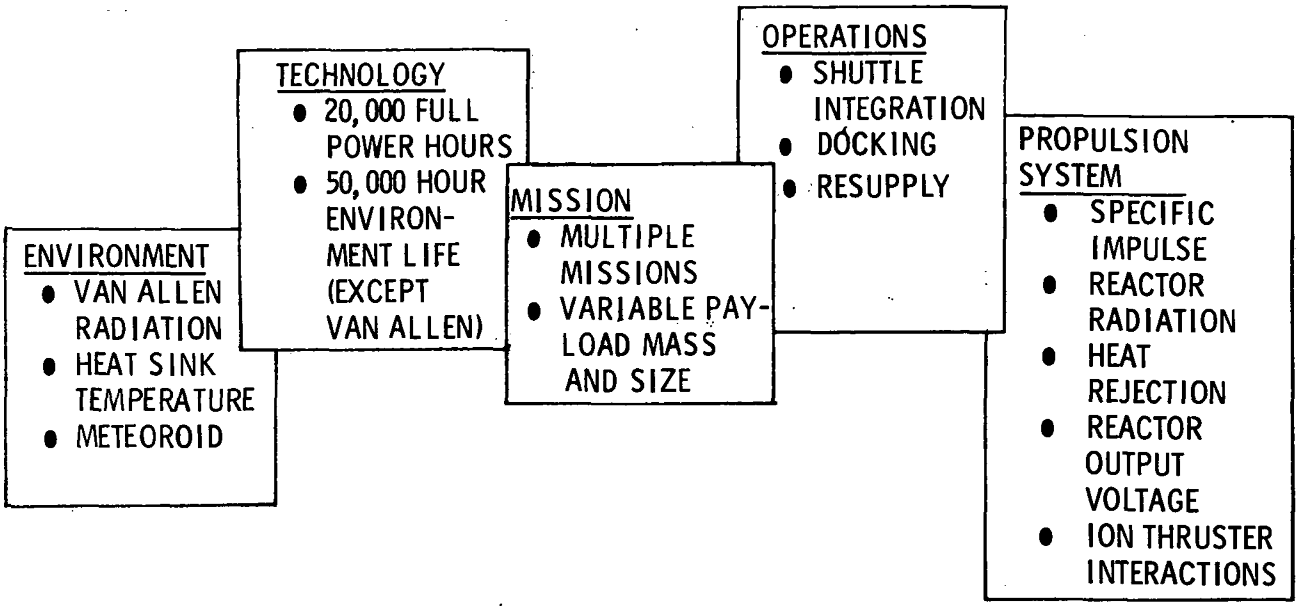


Figure 2-2. Key Configuration Drivers

2.2.1 ENVIRONMENT

2.2.1.1 Van Allen Radiation

The Van Allen radiation belt poses little or no problems for most space flights due to the relatively short time spent in the rather intense electron and proton radiation environment. A geocentric orbit mission mode which requires the NEP Stage, with its satellite payload, to spiral in and out of the Van Allen belt over time periods of several months, can result in deg-gradation of the power conditioning electronics unless electron and proton radiation shielding

is provided.* This radiation protection can be provided by a slight increase in the PC radiator thickness to allow fewer electrons and protons to penetrate the radiator panel and strike the susceptible electronics. Approximately the same shielding will be provided by either an aluminum or beryllium radiator panel; however, beryllium will impose the smallest weight penalty, and the highest cost penalty.

2.2.1.2 Heat Sink Temperature

For outer planet mission, the heat sink temperature is estimated to be approximately $166^{\circ}K$ ($-160^{\circ}F$), whereas, the heat sink temperature for the geocentric mission is estimated to be approximately $252^{\circ}K$ ($-5^{\circ}F$). This change in heat sink temperature will have a negligible effect on the high temperature ($\sim 1000^{\circ}$) primary radiator; however, the low temperature ($373^{\circ}K$) PC radiator must be sized for geocentric orbit application.

2.2.1.3 Meteoroid Environment

Meteoroid protection must be provided for the primary radiator to assure a 0.99 non-puncture probability in 50,000 hours. The near earth meteoroid flux model used in this study is contained in Appendix A.

Volkov (Reference 2-3) estimates that the interplanetary meteoroid flux is approximately 43 percent that of the near earth environment. However, recent data based on Pioneer 8 and 9 (and preliminary analysis of Peioneer 10 data) indicate that the interplanetary meteoroid environment may be equal to, or as much as a factor of 10 worse than the near earth environment.

The multi-mission NEP stage is designed to survive 50,000 hours in the near earth meteoroid environment.

*Electron and proton radiation protection may also be required for certain electronic components in the avionics module and the synchronous orbit payload. Solar arrays in the synchronous orbit payload are the most sensitive, and must be shielded to an equivalent integrated dose of 10^5 rads gamma, or less.

2.2.2 TECHNOLOGY

The NEP Stage propulsion system lifetime requirement is 20,000 full power hours. This applies primarily to the reactor and ion engines. All systems must meet a lifetime requirement of 50,000 hours in the operational space environment.

2.2.3 MISSION

The NEP Stage design objective provides full multi-mission capability for both interplanetary and geocentric earth orbit missions. Therefore, both types of mission environments must be evaluated to determine the most imposing design requirements placed on the stage. As an example, the primary heat rejection subsystem will be designed for the near earth heat sink temperature. Consequently, the heat rejection subsystem will be overdesigned for interplanetary missions.

The NEP Stage must be capable of transporting payloads of variable mass and size. Synchronous orbit payloads have been identified with masses of up to 2000 kg and dimensions up to 7.6 m long by 4.6 m in diameter. Payload mass not only affects trip time, but it can have a profound effect on the NEP stage design as well. The impact of payload mass on the NEP Stage design is of little or no significance if the thrusting is axial; however, if thrusting is perpendicular to the vehicle's major axis (as in a side thrust configuration), the potential center-of-thrust and center-of-gravity miss-match must be accounted for.

2.2.4 OPERATIONS

Since the NEP Stage is to be transported by the Space Shuttle, it must be designed to fit within the 4.6 m diameter by 18.3 m long Shuttle cargo bay (see Section 5). If the NEP Stage is to be launched with a payload and/or a chemical kick-stage, further constrints are placed on the size of the stage. A foldable or deployable configuration may facilitate Shuttle packaging.

An additional Shuttle integration constraint limits the location of the Shuttle payload center-of-gravity as shown in Figure2-3.

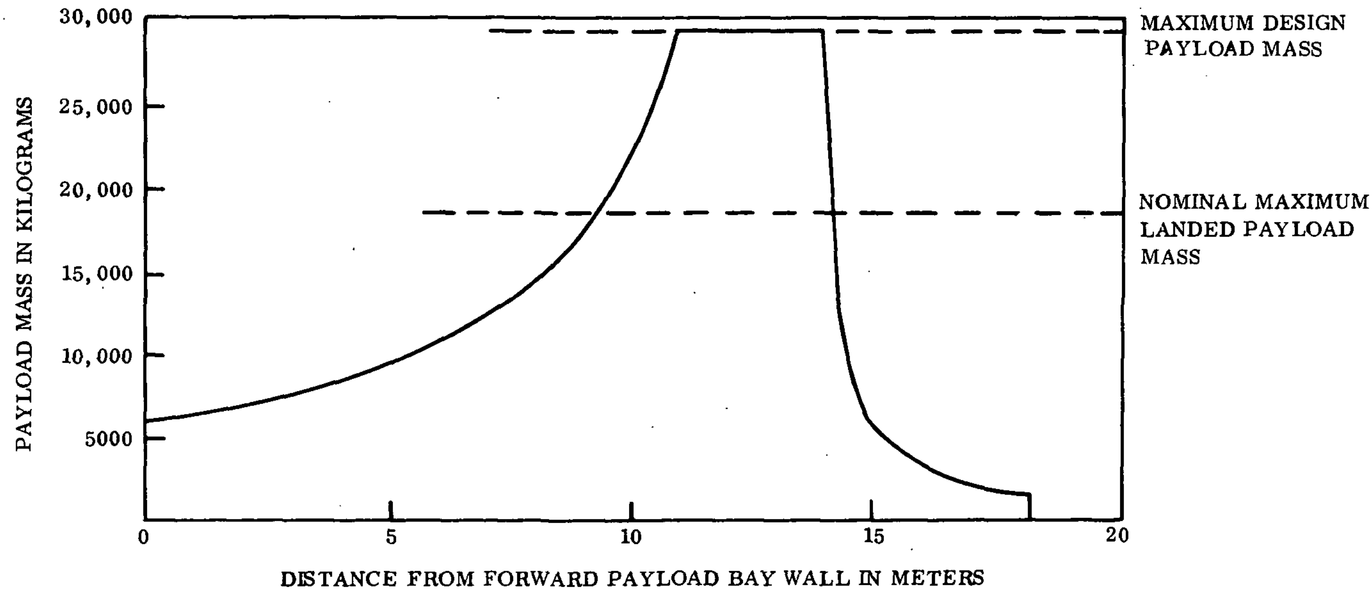


Figure 2-3. Payload Longitudinal Center-of-Gravity Limits

For geocentric orbit applications, the NEP Stage must be capable of docking and undocking with various payloads (passive and active). For this same mission mode, it reduces mission performance if the NEP Stage is required to carry enough fuel for the complete operational lifetime. Therefore, the NEP Stage performance is improved if resupply capability is provided for the mercury propellant, and other consumables expended during operation.

2.2.5 PROPULSION SYSTEM

2.2.5.1 Specific Impulse

Figure 2-4 shows the effect of specific impulse (I_{sp}) on the NEP Stage configuration and mission performance. Approximately a 20 percent reduction in trip time is achieved by going down in I_{sp} from 4000 sec to 3000 sec. However, this same reduction in specific impulse results in approximately a 55 percent increase in the required mercury propellant inventory. In addition, decreasing specific impulse from 4000 to 3000 seconds, results in an increase

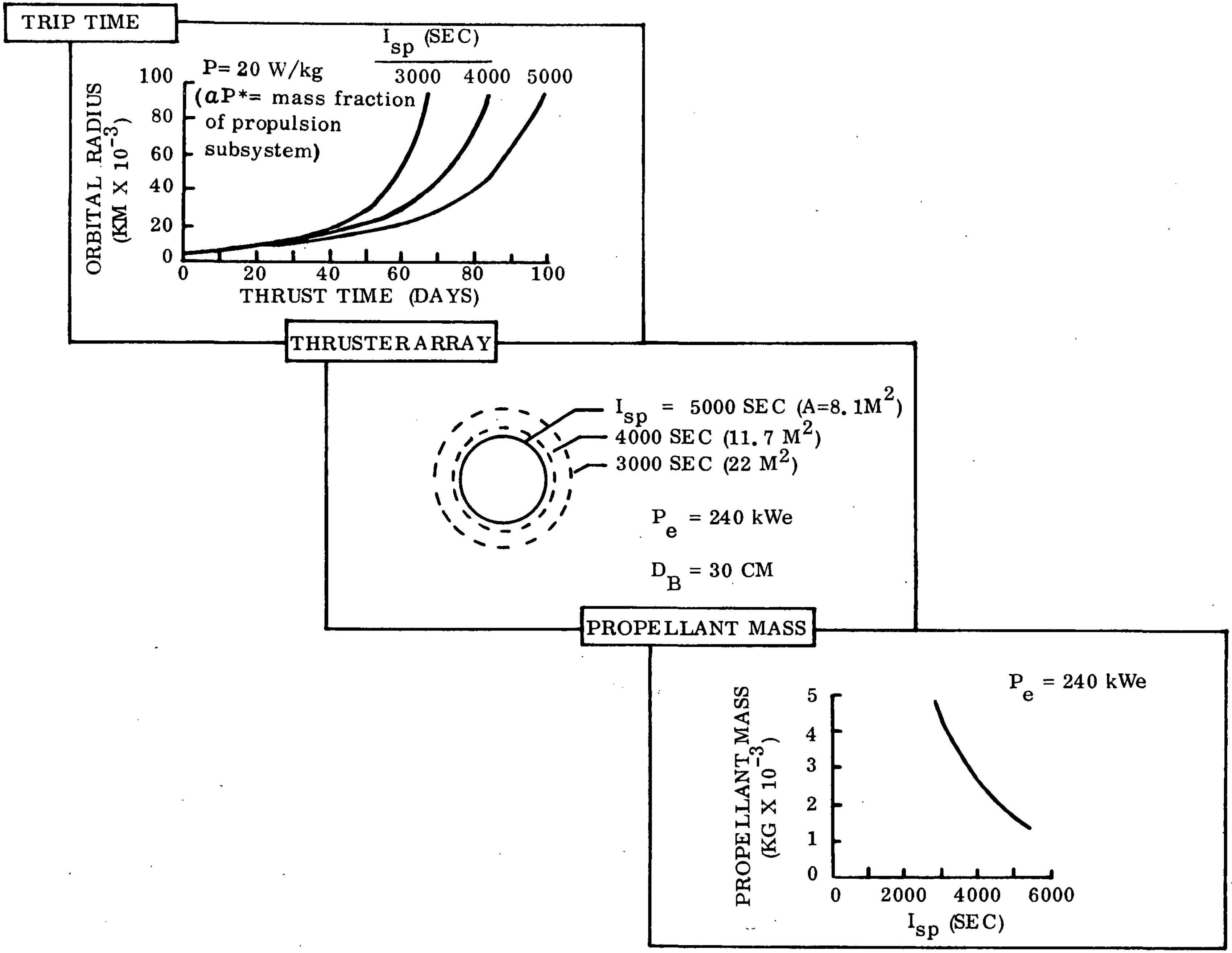


Figure 2-4. Effect of Specific Impulse on Mission Performance and NEP Stage Configuration

in the number of thrusters required, hence an increase in packaging area for the ion thruster array of approximately 90 percent.

2.2.5.2 Reactor Radiation

The location of the reactor is a key element in the design of a NEP system since it must be shielded from components that are susceptible to nuclear radiation (i.e., photovoltaic and semiconductor materials that may be in the NEP power conditioning, avionics module, and/or payload). Hence, the location of the reactor in relation to those components has a significant impact on the amount of neutron and gamma shileding required to reduce the overall cumulative mission dose at the nearest PC station to 10^{12} nvt (E_n > 1 Mev) and 10^6 rads gamma.

Present studies assume no rendezvous with the Space Shuttle. If future studies identify a requirement for the manned Shuttle to rendezvous directly with the NEP Stage, the impact of this mission operation on the shield weight and geometry remains to be defined.

2.2.5.3 Heat Rejection

The high temperature ($\sim 1000^{o}K$) primary radiator is sized to reject the thermal energy produced by the thermionic reactor that is not converted into useful electrical power. The passive low temperature ($\sim 373^{o}K$) power conditioning radiator is sized to reject the heat generated in the NEP Stage power conditioning modules. If the high temperature primary radiator is positioned next to the low temperature PC radiator, a thermal shield must be provided at the primary radiator/PC radiator interace.

2.2.5.4 Reactor Output Voltage

The reference thermionic reactor produces electrical power at approximately 23 Vdc. This electrical power is transmitted from the reactor to the power conditioners via low voltage power transmission cables. The greater the I^2R cable losses are, the higher the reactor operating power level must be to deliver a specified power level to the power conditioners. This increased power level results in a higher specific weight for the NEP Stage.

The I^2R losses that are generated in the low voltage cables must be dissipated to prevent conduction of this energy into the PC modules. The PC radiator area definition (and that of the primary radiator if the cables run across it) must allow for the heat rejection surface that is blocked by the low voltage cables.

2.2.5.5 Ion Thruster Interactions

Large thruster array areas can complicate the NEP stage design, and Shuttle packaging. For example, in a 240 kWe axial (end) thrusting configuration, a 3000 sec I_{sp} thruster array cannot be packaged in the Shuttle cargo bay without having the thrusters mounted on some type of foldable array. For a side thrusting configuration, if low voltage power transmission cables run the length of the large thruster array required for the example 3000 second system, the I^2R losses could significantly increase the specific weight.

The thrusters must be positioned such that they cannot act as a source of radiation scattering. For an example 3000 second I_{sp} system with axial thrusting, the large thruster array results in additional radiation shielding and higher weight to prevent significant scattered radiation.

The ion thrusters must be oriented such that mercury and sputtered grid material, such as molybdenum, are not exhausted over sensors and heat rejection surfaces which have emissive coatings which are subject to degradation. Where surfaces are exposed to the ion beam and the sputtered grid materials, protection will be necessary. Capton or titanium have the characteristics of light weight and low sputtering erosion. Shield thicknesses would be on the order of millimeters.

Data indicate that the problem of ion engine interactions with external vehicle surfaces is small at angles greater than 15 to 20 degrees from the thruster axis. (Reference 2-4). Surface deposition and increased surface absorptivity are negligible when the surface is at least 90 degrees to the thruster axis. Recent test data on an experimental 30 cm ion engine, with dished grid, indicate that the ion engine interactions with external vehicle surfaces may still be significant up to 25 to 30 degrees from the thruster axis (Reference 2-5). However, it is

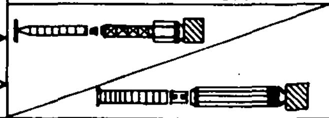
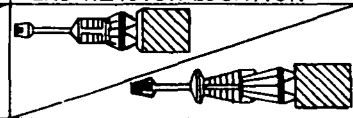
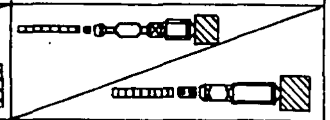
expected that future design efforts will improve the beam focusing characteristics of this 30 cm ion engine, resulting in a beam divergence comparable to that of the 20 cm ion engine.

2.3 CONFIGURATION ANALYSIS

A propulsion system configuration analysis was performed to arrive at an optimum NEP Stage design. Three families of NEP Stage propulsion system configurations were investigated: an end thrust (i.e., axial thrusting) configuration with a mid-reactor location, an end thrust configuration with the reactor(s) located at the end of the vehicle, and a side thrust (i.e., thrusting perpendicular to vehicle's major axis) configuration. Two different power levels were considered: one reactor delivering 120 kWe to the thrust subsystem, and two reactors (each at 120 kWe) delivering a total of 240 kWe to the thrust subsystem.

The configurations were rated in terms of four generic evaluation criteria: mission performance, versatility, development risk, and cost. The results of this analysis are summarized in Figure 2-5. The end thrust configuration, with end reactor location, is most attractive in terms of mission performance and overall operational versatility, because of its low specific mass, ease of Shuttle integration, and multi-mission (geocentric and interplanetary) capability. The side thrust configuration appears to be the most attractive in terms of development risk because of minimal ion engine and thermal interactions. No significant differences in development and production costs have been identified for the three propulsion system configurations.

The end thrust NEP Stage configuration with end reactor location, is the best suited configuration for combined geocentric and interplanetary missions. The reference end thrust NEP Stage design presented in the following section minimizes recognized potential ion engine and thermal interactions. The power level selected for the reference NEP Stage is 120 kWe, although higher power levels may result in improved performance in terms of reduced trip time or higher payloads. The maximum power level compatible with Shuttle integration for geocentric orbit applications is about 400 kWe.

		END THRUST MID REACTOR LOCATION	END THRUST END REACTOR LOCATION	SIDE THRUST
	120 kWe / 240 kWe			
MISSION PERFORMANCE	SPECIFIC MASS	3	1	2
	SHUTTLE INTEGRATION	2	1	1
VERSATILITY	MULTI-MISSION CAPABILITY	2	1	3
DEVELOPMENT RISK	RADIATION SCATTERING	3	1	2
	ION ENGINE INTERACTIONS	1	2	1
	THERMAL INTERACTIONS	2	3	1
COST	DEVELOPMENT	NO IDENTIFIED DIFFERENCES		
	PRODUCTION	NO IDENTIFIED DIFFERENCES		

1 - MOST ATTRACTIVE
2 - ATTRACTIVE
3 - LEAST ATTRACTIVE

- PRELIMINARY ASSESSMENT INDICATES THAT END THRUST DESIGN WITH END REACTOR LOCATION BEST FOR BOTH GEOCENTRIC ORBIT AND INTERPLANETARY MISSIONS.

Figure 2-5. Configuration Analysis NEP Stage Propulsion System

SECTION 3

REFERENCE NEP STAGE DESCRIPTION

The reference NEP Stage is basically a conical configuration with a cylindrical heat pipe primary radiator. The reactor is boomed to minimize shielding weight and ion engine interactions with minimum low voltage cable losses. An array of 30 cm mercury electron bombardment ion engine provides axial thrust at a specific impulse of 4000 sec. The thruster array is composed of 24 engines, including 20 percent spares.

3.1 NEP STAGE DESIGN AND PERFORMANCE SUMMARY

The conceptual design of the reference 120 kWe NEP Stage showing the general arrangement of the vehicle (in-flight configuration) is presented in Figure 3-1. A summary of the major performance parameters is shown in Table 3-1. The overall dimensions of the basic NEP Stage are 12.8 m long to the aft end of the avionics package with a maximum diameter of 4.6 m.

The major NEP Stage system masses are:

1.	Power Subsystem	3030 kg	
2.	Thrust Subsystem	755 kg	
3.	Propellant Subsystem	5740 kg	(Typical - planetary missions)
4.	Avionics Subsystem	460 kg	

The specific mass of the reference NEP Stage is 32 kg/kWe (does not include the propellant or avionics subsystems) based on 120 kWe net power delivered to the thrust subsystem. Inclusion of the avionics subsystem would add 3 to 4 kg/kWe to the stage specific mass. As shown by the power balance diagram of Figure 3-2, this 120 kWe consists of a total of 114 kWe to the main power conditioning modules and 6 kWe to the specific ion engine power conditioning. Approximately 110 kWe of this power is delivered to the ion engines in the form required to produce electric propulsion. In order to provide the 120 kWe to the thrust

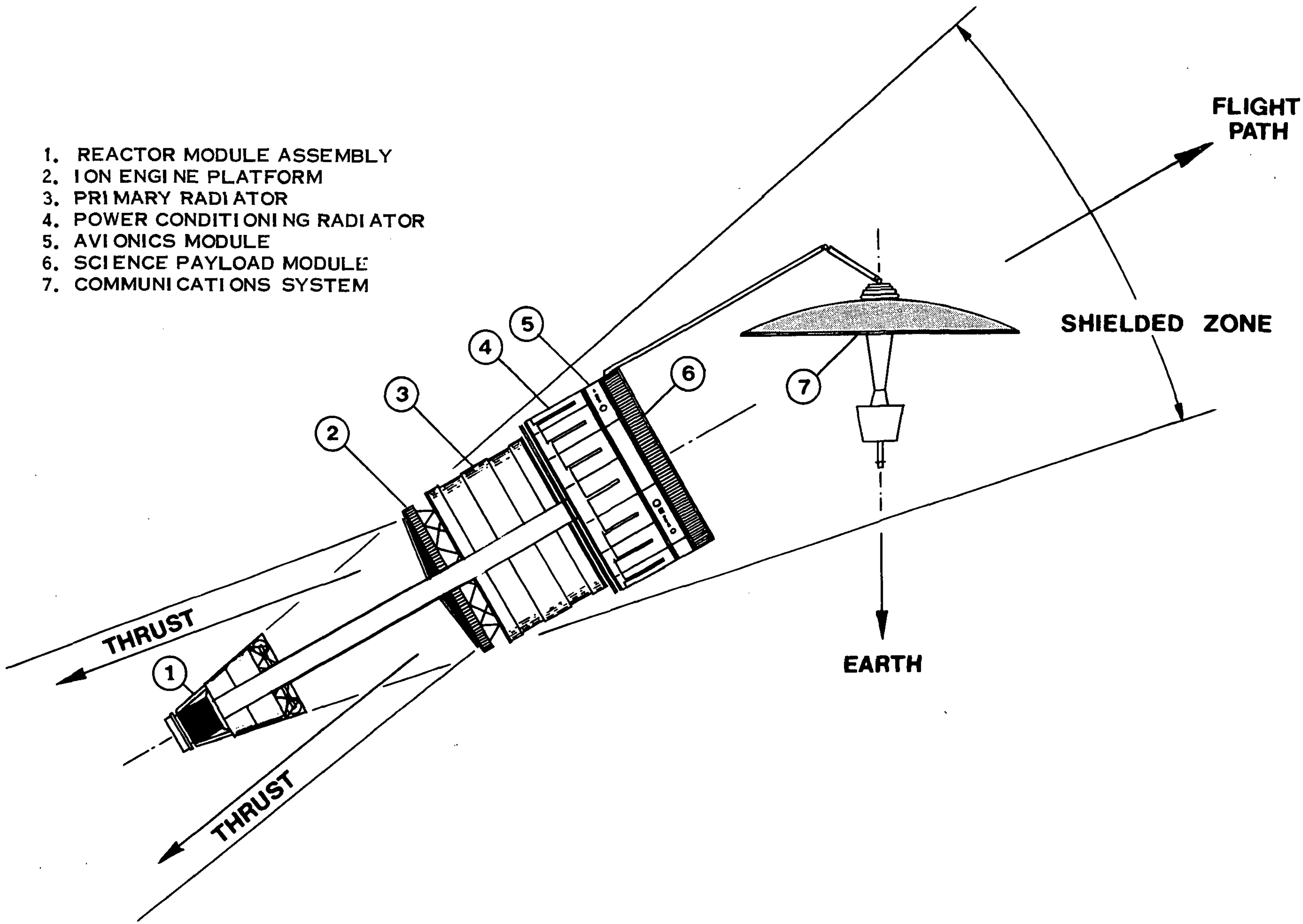


Figure 3-1. General Arrangement of 120 kWe NEP Stage End Thrust Configuration

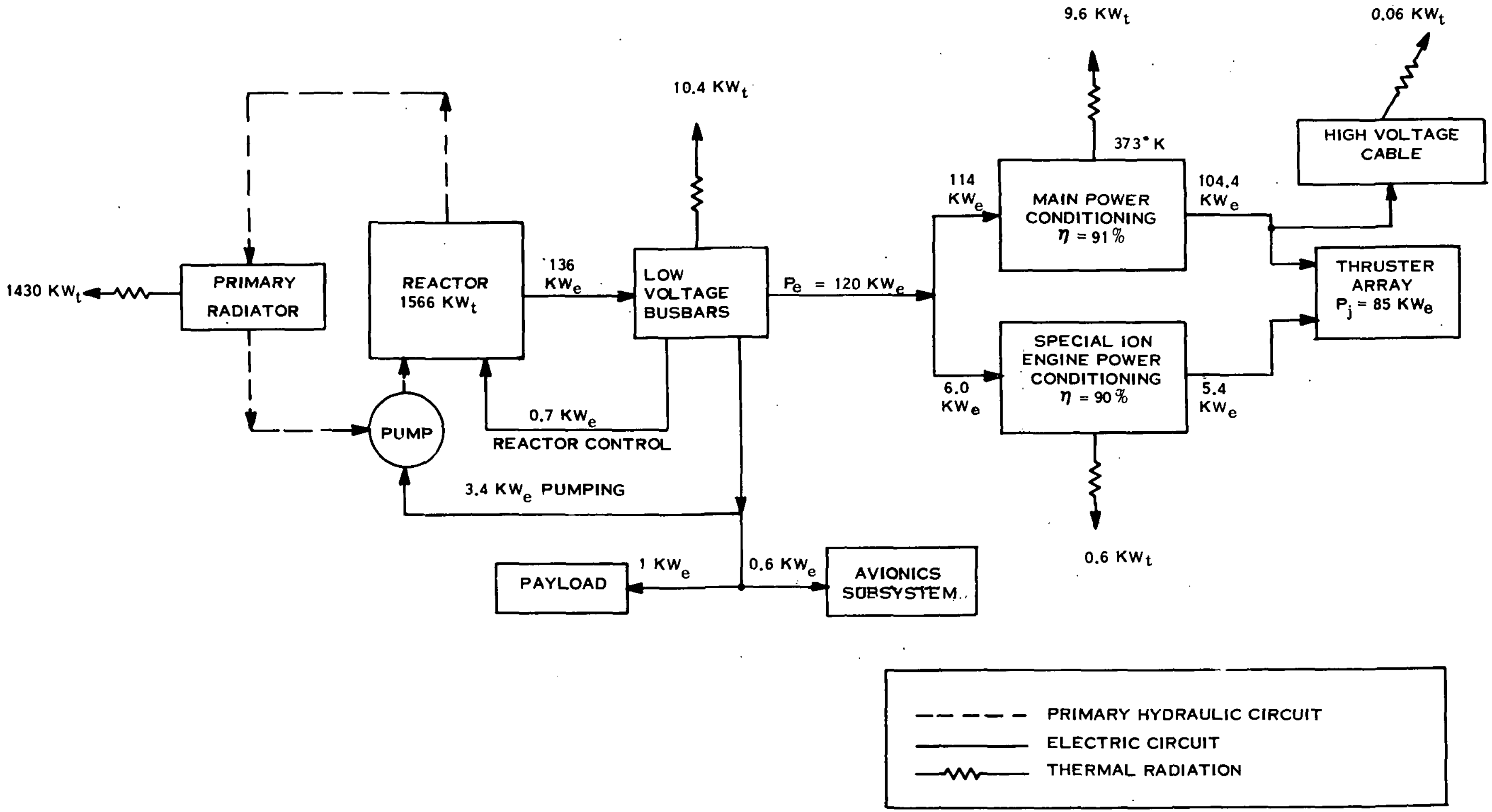


Figure 3-2. 120 kWe NEP Stage Power Balance and Distribution (End of Mission)

subsystem, the reactor generates 1580 kWt converting 136 kW to electrical power and rejecting the rest via the primary radiator. The 16 kWe representing difference between the reactor output and the thrust subsystem input provides the electrical power for the operation of the powerplant and payload, and the various losses in the electrical circuit. Approximately 115 kWe of the 136 kWe total represents useful electrical loads throughout the spacecraft with the remaining 21 kWe representing losses in the power conditioning and cables.

3.2 NEP STAGE COMPONENT SUMMARY

Starting at the aft end of the NEP Stage and proceeding along the vehicle length, the components are arranged in the order discussed below.

Table 3-1. Key Performance Parameters of Baseline 120 kWe NEP Stage

Power Level to Thrust Subsystem	120 kWe @23 volts
Specific Impulse	4000 sec.
Propulsion Efficiency	71%
Specific Mass (does not include Propellant or Avionics Subsystem)	32 kg/kWe
Power Available to Avionics Subsystem	1 kWe
Full Power Operational Life	20,000 hours
Total Orbital Life	50,000 hours
NEP Stage Mass	4845 kg
Dimensions (Stowed)	12.8 m (1) 4.6 m (max. diam.)
Engine Restart Capability	Yes
Propellant Tank and Hg Feed System	3% of propellant mass
Engine Type	30 cm Hg ion
Number of Thrusters	24

The thermionic reactor, 0.86 m long and 0.71 m in diameter, is joined to the shield by a conical structure. A pumped primary loop (NaK-78 coolant) in conjunction with a heat pipe radiator constitutes the heat rejection subsystem. The 44 cm thick LiH neutron shield, next in line, is conical in shape, with a mean diameter of 1.5 m. This shield reduces the integrated mission neutron dose to the 10^{12} nvt ($E_N > 1$ MeV) at the power conditioners. The total neutron shield weight is approximately 460 kg.

The mercury propellant tank is 0.36 m in axial thickness, and is located forward of the LiH neutron shield. The stored mercury serves as the primary gamma shield, reducing the integrated mission photon dose to 10^6 rads at the power conditioning electronics. Further evaluation is required to determine if the mercury propellant must have its own heat rejection subsystem to maintain the temperature of the mercury below its boiling point of approximately 600^{o}K. A heat pipe system can be employed if direct radiation to space does not maintain an allowable mercury temperature.

The forward base of the reactor/shield assembly has a Kapton or fiberglass material on its surface for protection from surface erosion and degradation due to interaction with the ion engine exhaust. A titanium clad surface could also be employed. Bus bars in the form of aluminum cables (copper near the hot reactor) carry the reactor electrical output across the reactor shielding, down the length of the stage, to the power conditioning modules.

A titanium-clad cylindrical structure (0.152 cm wall thickness) houses the NaK coolant lines to the primary radiator, and the mercury feed lines that supply propellant to the ion thrusters. This structure is required to support the reactor-shield assembly, and to protect part of the primary coolant loop and the mercury feed lines from ion thruster interactions. Meteoroid protection is also provided by this structure. The aluminum low voltage cables are housed in a trough assembly, external to the titanium-clad structure, which also provides shielding from the ion thruster exhaust. The cables are electrically insulated on one side, and radiate I^2R losses to space on the other side.

Next in line is the ion thruster array which contains twenty-four, 30 cm mercury ion engines. This number includes 20 percent redundancy.* The mercury ion engines are canted out at an angle of nine degrees (results in a one percent loss of effective thrust) to reduce mercury impingement degradation. This angle in combination with the cone angle of the reactor/shield assembly, will result in little or no mercury impingement degradation of the reactor/shield and its support structure. The thrusters are mounted in two clusters allowing for two zones on either side of the vehicle, free from mercury impingement, where the low voltage cables are located.

Approximately twelve of the ion engines can be gimbaled to provide for roll thrust vector control about the thrust axis and yaw control. The ion engine spacing permits rotation of the gimbaled ion engines $\pm$ 10 degrees. Pitch (and yaw) control can be achieved by mounting the thruster array on hinged panels. This allows the effective thrust angle to be increased more on one side than on the other, resulting in a thrust differential. This may also be accomplished by decreasing the thrust level of the engines on one side.

The maximum allowable ion engine temperature is $523^{\circ}K$. To maintain this temperature, each engine must reject approximately 500 watts. This heat rejection can be provided by approximately three square meters of surface, located forward of the thruster array.

A multi-foil thermal shield is located at the interface between the thruster array and the primary radiator to thermally separate the high temperature radiator and low temperature thruster array.

The primary radiator and supporting structure comprise the next NEP Stage section. The primary radiator is cylindrical, having a length of 2.6 m with approximately 30 m^2 of surface area. It is formed of ~ 700 sodium filled heat pipes which are brazed to circumferential stainless steel headers off the primary NaK loop. The cylindrical radiator configuration was selected to facilitate manufacturing. If length becomes an important consideration, a conical radiator will result in the minimum length vehicle for Shuttle packaging. The low voltage

*Previous studies assumed that during thrusting, 20 percent of the ion engines were not operating and served as spare engines if any of the operating engines failed. The most recent approach to redundancy is to have all the ion engines operating at reduced power and if an engine fails, the remaining engines are all operated at a slightly higher power level.

cables that run the length of the radiator are thermally insulated from the high temperature surfaces. This limits the maximum allowable cable temperature to $370^{o}K$.

A preliminary analysis was performed to determine what the activation level of the coolant in the primary loop in terms of radiation dose to the power conditioners. It is estimated that for a 120 kWe system, the equilibrium coolant activation level in the primary loop is approximately 270 Ci (after 50 hours of operation). Based on this activation level, the resultant radiation dose contribution is negligible compared to that from the reactor itself, if the maximum allowable integrated mission gamma dose is 10^{7} rads. However, if the maximum acceptable limit is 10^{6} rads over the entire mission, the integrated gamma dose from the unshielded Na-24 in the primary loop to the power conditioning electronics, becomes more significant and warrants closer examination. A heat exchanger and secondary coolant loop (approximately 1 to 2 kg/kWe weight penalty) will eliminate this potential problem.

The next NEP Stage section contains the power conditioning (PC) radiator. A thermal shield located at the primary radiator/PC radiator interface thermally separates the high temperature (approximately $1000^{o}K$) and low temperature ($373^{o}K$) components. The conical PC radiator is sixteen-sided in cross-section, and is 1.55 m long, with 20.8 m^{2} of surface area. Individual power conditioning modules (one for each ion engine plus approximately four for hotel loads) are mounted to the inner surface of the 0.38 cm thick beryllium radiator panels. The function of the PC radiator is to limit operating temperatures to a maximum of $373^{o}K$ in the power conditioning modules by dissipating the heat generated in the modules via direct radiation to space. The individual PC modules operate at an efficiency of approximately 91 percent.

Mission trip times through the Van Allen radiation belt could require electron and proton radiation protection, in addition to that already afforded by the 0.38 cm of beryllium. In this event, the individual PC modules can be housed in beryllium structures to increase the effective thickness that the electrons and protons must penetrate before reaching the power conditioning electronics.

High voltage transmission lines, electrically insulated aluminum cables, transport the 2000-volt electrical power from the power conditioners to the ion thrusters. The low electrical power losses of approximately 50 watts permit the high voltage cables to be located on the inner surface of the PC radiator panel. The high voltage cables must be thermally insulated from the high temperature surfaces of the primary radiator.

The foremost section of the NEP Stage contains the avionics subsystem and the payload docking structure. A detailed mass summary of the 120 kWe NEP Stage is presented in Table 3-2.

As previously described in Section 2.1, four basic subsystems make up the NEP Stage: the power subsystem, the thrust subsystem, the propellant subsystem, and the avionics subsystem. A description of each of these subsystems is presented in the following sections.

3.3 POWER SUBSYSTEM

The power subsystem is made up of all the components needed to generate the NEP Stage electrical power and all the shielding components necessary to protect the nuclear radiation sensitive components. The principal parts of the NEP Stage power subsystem are a thermionic reactor, a main heat rejection subsystem including radiator, EM pumps and associated plumbing, radiation shielding, a power conditioning and electrical distribution subsystem for the EM pumps and reactor control actuators, low voltage power transmission cables, a startup auxiliary power supply, and related structure.

3.3.1 REACTOR

The reference multi-mission NEP Stage employs an internal fuel thermionic reactor that provides 120 kWe at approximately 23 volts (dc) to the thrust subsystem at End of Mission (EOM). A brief summarization of the dimensions and conditions of the U-235 fueled thermionic reactor operating at $\sim$23 volts and 136 kWe (gross) output are given in Table 3-3.

Table 3-2. 120 kWe NEP Stage Mass Summary
(End Thrust Configuration)

Power Subsystem

Component	Mass - kg
Reactor	1440
Heat Rejection Subsystem	650
Neutron Shield	460
Hotel PC	50
Hotel PC Radiator	20
Low Voltage Power Transmission Cables	260
Startup Auxiliary Power Supply	50
Structure	90
Total	3030

Thrust Subsystem

Component	Mass - kg
Thruster Array	305
PC Modules	300
PC Radiator	125
High Voltage Power Transmission Cables	1
Structure	25
Total	755

Propellant Subsystem

Component	Mass - kg
Mercury Propellant	5500 - Interplanetary 2360 - Geocentric
Tanks and Distribution	165
Total	5740 - Interplanetary 2600 - Geocentric

Avionics Subsystem

Component	Mass - kg
Attitude Control	34
Flight Command/Structure	289
Flight Telemetry	62
Video/Lighting	15
Docking	35
Thermal	25
Total	460

Total NEP Stage Mass (does not include Hg Propellant)	3950 kg

Table 3-3. Reactor Characteristics Reference 23-Volt Flashlight Reactor Spacecraft

Configuration Parameters	
Diameter, m	0.711
Length, m	0.864
Mass, kg	1440
Number of Thermionic Fuel Elements	162
Number of Independent Circuits	27
Design Point Performance	
Thermal Power at EOM, kWt	1566
Electrical Power at EOM, kWe	136
Electrical Power Voltage, Volts	22
Coolant Temperature Rise at EOM, ^oK	95
Average Emitter Temperature at EOM, ^oK	1885
Coolant Pressure Drop, N/m^2	9130

3.3.2 RADIATION SHIELDING

In accordance with the established guidelines for this study, the power conditioning electronics and other radiation sensitive components have been shielded to neutron and gamma integrated dose limits of 10^{12} nvt ($E_n > 1$ MeV) and 10^6 rads, respectively. Neutron and gamma shield designs are based on analyses conducted by Oak Ridge National Laboratory (Reference 3-1), and more recent analyses being performed by NASA/LeRC.

3.3.2.1 Neutron Shield

The neutron shield consists of a lithium hydride stainless steel honeycomb enclosed in a stainless steel can. The lithium hydride performs most of the required neutron shielding with additional neutron attenuation contributed by the mercury propellant. The neutron shielding requirement of 10^{12} nvt ($E_n > 1$ MeV) is satisfied with about 44 cm of lithium hydride. However, calculations being performed by NASA/LeRC indicate that approximately 55 cm of LiH is required. Based on a 44 cm LiH neutron shield thickness, the neutron shield subsystem is composed of 450 kg of lithium hydride and 10 kg of stainless steel, about three percent of the lithium hydride by volume.

It is estimated that no auxiliary active cooling of the LiH shield is required in order to maintain the shield material temperature below an allowable 755 ^{o}K. Heat is conducted from the reactor face of the shield by the lithium hydride and stainless steel components to the outer surface of the shield where it is radiated directly to space.

3.3.2.2 Gamma Shield

The primary reactor gamma shielding for the NEP Stage is provided by the liquid mercury propellant located in a 1.8 m diameter by 0.67 m long cylindrical tank. The axial mercury thickness provided by this tank geometry reduces the overall mission integrated dose to the PC electronics to 10^6 rads for the 20,000 hours full power mission. To reduce this integrated dose to 10^5 rads, about one centimeter of permanent tungsten gamma shielding is required.

3.3.2.3 Heat Rejection Subsystem

The primary heat rejection subsystem is comprised of:

1. Primary heat pipe radiator
2. EM pumps
3. Accumulators
4. Piping
5. NaK coolant

The primary radiator consists of ~700 sodium filled stainless steel heat pipes axially mounted on stainless steel circumferential headers. Each heat pipe is designed to reject approximately 2 kWe of waste heat. The heat pipes are rigidly joined and brazed to the circumferential headers that run off the primary NaK coolant duct to form a cylindrical radiator surface of 4.0 m in diameter. The total surface area of the radiator is 31.9 m^2, its length is 2.6 m, and its mass including headers and associated coolant is 359 kg. The surface area of the primary radiator allows for the area blocked by the power transmission cable that runs its length.

The primary radiator is designed so that at the end of the 50,000 hour NEP Stage lifetime, 90 percent of the heat pipes survive the meteroid environment. To accomplish this, the individual heat pipes are designed to have a 91 percent probability of survival (Reference 3-2). Exposed sections of the primary loop are double-walled for meteoroid protection. The heat pipes serve as meteoroid "bumpers" to protect the circumferential headers and primary ducting that are internal to the primary radiator. The radiator is designed to reduce the temperature drop between the coolant loop and the heat pipe panel to approximately 25°K.

Two AC induction EM pumps in series provide the coolant circulation for the heat rejection loop. Each unit weighs 50 kg. Required cooling of the electrical coils is accomplished passively with multiple heat pipes which radiate directly to space. The efficiency of the pumps is 15 percent, but only 60 percent of the waste heat generated by the pump inefficiency is dissipated to space, with the remainder deposited in the pumped coolant.

Four accumulator tanks, two active with gas pressurized bellows and two passive spherical tanks, provide for coolant expansion and pressurization. The accumulators each weigh approximately 13 kg fully charged with coolant.

Feed line piping of approximately 9.7 cm inner diameter and 0.13 cm wall thickness transports the heat rejection loop coolant from reactor to the primary radiator and back to the pumps and the reactor. The total weight of this piping, including coolant, is 227 kg.

3.3.3 ELECTRICAL SUBSYSTEM

The power system electrical subsystem includes the hotel power conditioning equipment for the EM pumps along with the associated cooling radiator, power cabling to the pumps and reactor control actuators, and the low voltage power transmission cables from the reactor to the main power conditioning modules.

The hotel power conditioning is based on similar components designed for the 240 kWe Thermionic Spacecraft Study of Contract No. JPL 952381 (Reference 3-3). The pump power conditioning is performed by the main power conditioning modules described in Section 3.2.2.

However, the portion of the total PC mass which is attributable to the conditioning of the EM pump power has been estimated and tabulated separately. The hotel PC weighs approximately 50 kg and supplies 3.4 kWe of variable frequency AC power to the EM pump at a conversion efficiency of 90 percent. A total of 20 kg of PC radiator dissipates the waste heat generated by the pump power conditioning. The medium voltage cable transporting the electrical power to the pump weighs approximately 1 kg and generates only 20 watts of resistive power loss. The power cable to the reactor control actuators is negligible in mass and power loss.

A segmented transmission line of copper cable, aluminum bus bar and aluminum cable carries the ~ 23 volt power from the reactor to the main power conditioning modules. The total mass of the low voltage circuitry is 260 kg and the resistive losses total 10 kWe. Sufficiently exposed cable surface area is provided so that all the low voltage cable losses , plus heat conducted from the reactor can be rejected by radiation to the space. The temperature of these cables adjacent to the main power conditioning modules is maintained at 373°K.

3.3.4 STARTUP AUXILIARY POWER SUPPLY

The startup auxiliary power supply consists of nickel-cadmium batteries which provide electrical power for the reactor startup operations and for coolant circulation following reactor shutdown. The masses and volumes for this battery matrix are shown in Table 3-4.

In order to accommodate an arbitrarily assumed 200-watt avionics subsystem load, and assuming an allowable 60 percent depth-of-discharge condition, about 610 watt-hours of battery capacity must be provided.

A low flow pump circulation will be necessary to minimize temperature variations within the coolant loop before reactor start-up. Approximately 30 minutes after leaving the Shuttle orbiter, the reactor start-up sequence is initiated, during which time cool-down takes place, with the potential for freezing. Allowing a five percent pump flow, approximately 10 watt-hours of battery energy are necessary for pre-start circulation.

Table 3-4. NEP System Start-Up Battery Matrix**

Operation	Mass Kilograms	Volume Meters^3
Reactor Start-up, Including 200 Watt Net Spacecraft Load*	29.5	0.0067
Pre-start Circulation	0.45	0.0002
Shutdown Circulation		
Near Earth (38.9 hours)	18.1	0.0043
Deep Space (7.8) hours)	3.6	0.0009
Battery Charge Regulator	1.4	0.0016

*Net spacecraft load of 200 watts is assumed arbitrary.
**Nickel-cadmium cell construction.

Similar to pre-start, coolant must be circulated following reactor shutdown (assumed to be inadvertant) to prevent radiator freezing and to dissipate heat from the fission product decay. Near Earth, 400 watt-hours of energy are required, and 80 watt-hours are required in deep space.

Supplying 200 watts (e) to the avionics subsystem, the total APS battery weight is about 50 kg, including a maximum of 18.1 kg for near Earth coolant circulation in the event of inadvertent shutdown, and 1.4 kg for the battery charge regulator, and 0.45 kg for pre-start circulation.

3.3.5 STRUCTURE

Power subsystem structural elements are required in three general areas:

1. Support and attachment members connecting the reactor, radiation shield, and heat rejection components.

2. Strengthening rings, etc., for the primary radiator.

3. Guiderail structure for separation of the Centaur kick-stage from the NEP Stage after high energy earth escape.

The total mass of this structure is estimated at 90 kg.

3.4 THRUST SUBSYSTEM

The thrust subsystem contains all the components and subsystems needed to convert the raw power generated by the power subsystem into ion thrust for NEP Stage propulsion. The major components that form the thrust subsystem are :

1. The ion engines
2. The main power conditioning modules (one per ion engine)
3. The special ion engine power conditioning modules (one per ion engine)
4. A passive PC radiator
5. High voltage power transmission cables between the PC modules and the ion engines
6. The structural members for each of the components

3.4.1 ION ENGINES

The ion engine subsystem consists of 24 thrusters. This number includes 5 spare engines which can be held in a stand-by mode, or can be operating at reduced power with the other engines. The thruster array is based upon the hardware and analytical techniques being developed for solar electric propulsion (Reference 3-4). Twelve of the engines are gimbaled to provide for roll control about the thrust axis and yaw control. The ion engine spacing permits rotation of the gimbaled ion engines $\pm$ 10 degrees. Pitch (and yaw) control can be achieved by mounting the thruster array on hinged panels which allows the effective thrust angle to be increased more on one side than on the other, resulting in a thrust differential, or by decreasing the thrust level of the engines on one side.

3.4.2 MAIN POWER CONDITIONING

The main power conditioning design is based on components similar to those defined in previous spacecraft design studies (Reference 2-2). Power is delivered from the reactor leads at a potential of approximately 23 volts and is distributed to the power converters. The 27 converters (one for each of the 6 TFE units) change the low voltage DC output of the thermionic reactor to AC and transform the ~23 volt reactor output to ~2000 volts for use by the main power conditioners for the ion engines. With individual power conditioners for each thruster, compensation for engine arcing is provided within the control circuit of each conditioner. Some of the ~ 23 volt input to the inverters is transformed to ~50 volts for input to the auxiliary hotel power conditioner.

Two different schemes for the main power conditioning are currently being evaluated. The main difference between the two approaches is reliability and the method of distribution of the beam power to the ion engine power conditioners. Both systems operate at an overall efficiency of approximately 91 percent. Further analysis and evaluation is required to determine the preferred approach.

3.4.3 SPECIAL ION ENGINE POWER CONDITIONING

The special power conditioning modules provide for all ion engine electrical loads, except for the high voltage screen supply. These other loads amount to 5 percent of the total power for each ion engine, about 5.4 kWe total in this application. The ion engines require a total power input of 6 kWe, assuming a 90 percent efficiency for these modules. The weight of all 24 units, which includes spares for the five spare ion engines, is estimated at 300 kg. These 24 special ion engine PC modules are located on the main PC radiator.

3.4.4 POWER CONDITIONING RADIATOR

The function of the power conditioning radiator is to maintain desired operating temperatures of $373^{o}K$ in the power conditioning modules by dissipating the heat generated in the modules via direct radiation to space. The design of the radiator is based on the results of the 240 kWe Thermionic Spacecraft Study performed by General Electric under Contract No. JPL 952381 (Reference 3-3).

The radiator is sixteen-sided in cross-section and is formed from individual conduction fin panels, approximately 0.38 cm thick, having a total surface area of 20.8 m^2. This total radiator area includes the redundant radiator area for the 5 spare power conditioning modules and the various hotel power conditioners. The overall length of the PC radiator is 1.5 m and weighs 125 kg.

The power conditioning modules are distributed uniformly around the periphery of the radiator at eight circumferential locations. Two flat panels cool the components of each module. The low voltage transmission cables extend down the flat sides of the radiator in the axial direction and consequently cover portions of the radiator panel surfaces.

3.4.5 HIGH VOLTAGE POWER TRANSMISSION CABLES

The high voltage transmission lines are electrically insulated aluminum cables which transport electrical power at ~ 2000 volts from each power conditioning module to its corresponding ion engine. The total mass of all the high voltage cables is only one kilogram. Insulation-support requirements, which were not evaluated in detail, would add 5 to 10 kilograms to this mass. The low electrical losses of 60 watts allows the cables to be located along the inside surface of the PC radiator panels. The cables are thermally insulated from the high temperature primary radiator by multi-foil insulation placed in a trough.

3.4.6 STRUCTURE

The thrust subsystem structural requirements consist of the following:

1. Support and attachment members for the ion engine thruster array.
2. Support and attachment members for the power conditioning modules and power conditioning radiator.
3. Docking assembly
4. Guide-rail structure for separation of the Centaur kick-stage from the NEP Stage after high energy earth escape.

The total mass of all this structure is estimated at 25 kg.

3.5 PROPELLANT SUBSYSTEM

The propellant subsystem consists of the mercury propellant, its cylindrical containment tank, and the propellant distribution system.

The tank design provides for positive mercury expulsion via a metal bellows system pressurized by a cold gas system. This also assures that no voids will form in the tanks during a mission coast phase, which, if incurred, would result in radiation streaming.

The mercury propellant tank is sized to contain 5500 kg of mercury, that required for a 20,000 full power hour mission. The mass of the propellant storage tanks and the propellant feed system is estimated at three percent of the maximum propellant mass, or 165 kg. The details of the propellant feed system have not been investigated.

The mass of the propellant subsystem is not included in the specific mass of the NEP Stage.

3.6 AVIONICS SUBSYSTEM

This section describes the avionics subsystem which serves as the command and control module of the NEP Stage. The avionics subsystem includes all of the components and subsystems necessary for mission operations other than the nuclear reactor power generation system and the electric propulsion system used to meet basic mission impulse requirements. The major subsystems include attitude control, flight command, flight telemetry, video/lighting, docking, and thermal control. Figure 3-3 shows the location of the avionics subsystem with respect to all the NEP Stage subsystems and the payload.

A geosynchronous earth orbit mission is assumed for the design of the avionics subsystem. Many of the components of the avionics subsystem will be directly applicable to interplanetary missions. Major differences will be in the selection of attitude control sensors, implementation of data handling hardware, software for the Thrust Vector Control (TVC), communication requirements, and certain components for functions peculiar to the geosynchronous orbit mission.

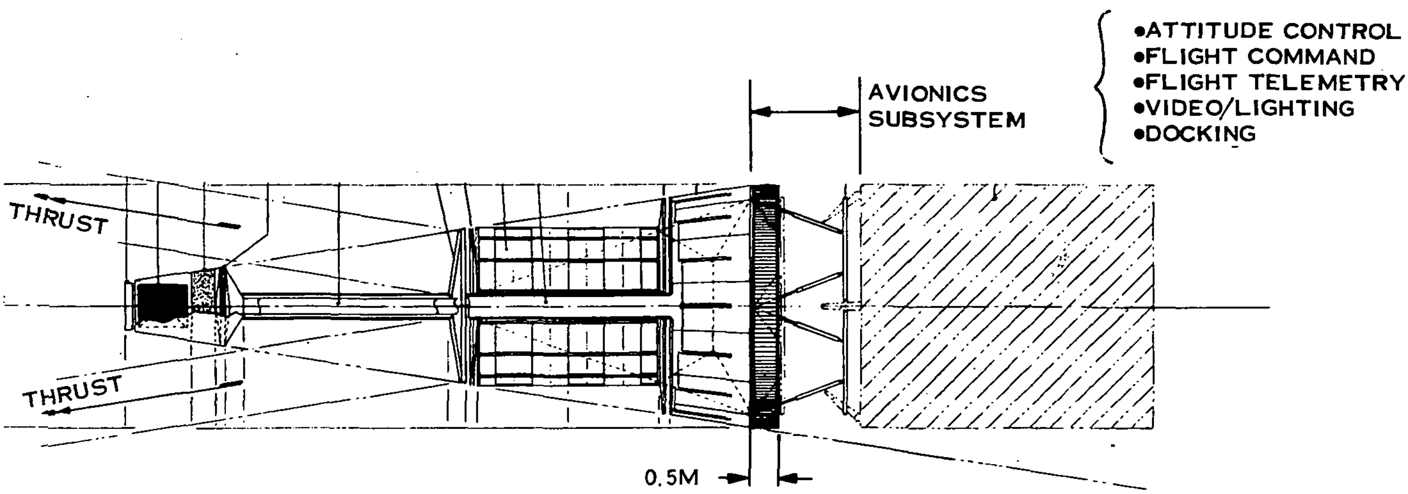


Figure 3-3. Avionics Subsystem Location

The reference mission consists of three phases:

1. A 1983 Shuttle launch, a NEP system consisting of a NEP Stage and a Propellant Logistics Depot (used for in-orbit refueling purposes) into a 435 km low earth orbit

2. A deployment transfer orbit phase for transport and deployment of the PLD by the NEP Stage into an intermediate elliptical parking orbit (14,800 km x 35,800 km, 15^{o} inclination)

3. An operational transfer orbit phase in which the NEP Stage transports payloads between the intermediate parking orbit and the synchronous equatorial orbit at an altitude of 35,800 km (see Section 4.2 for details of the geosynchronous orbit mission).

The approach taken for the design of the avionics subsystem is depicted in Figure 3-4. The approach is quite straightforward, although the conceptual nature of this effort did not permit definition of requirements to sufficient detail to specify more than the first level of design. In cases which these requirements were particularly difficult to define, such as in the impulse requirements for the auxiliary propulsion system, values which are considered conservative were arbitrarily established.

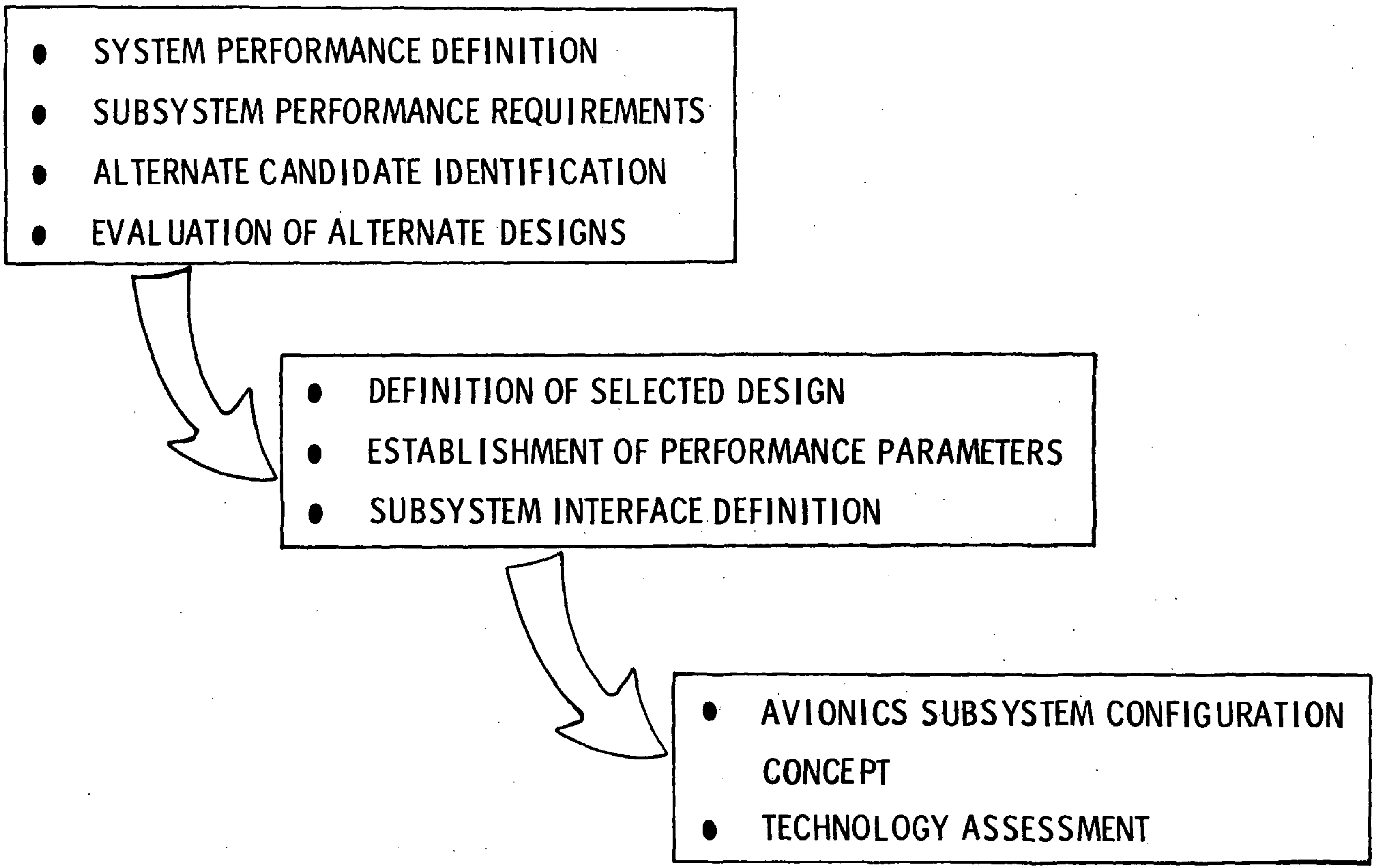


Figure 3-4. Avionics Subsystem Design Approach

The basic performance requirements for the various phases involve considerable trajectory analysis in the establishment of total ΔV, thrust vector orientation description, and the sensor positioning on the NEP configuration required to obtain the necessary sensor information. The final concept taken for the avionics subsystem design is to define a system in which the sensing and control components are satisfactory for the most demanding function and the propulsion functions are accommodated by the primary thruster system, thus lessening the impact on the avionics subsystem design.

Alternate approaches for design of certain subsystems are identified, trade studies are presented, and a preferred design approach selected.

Block diagrams and component performance parameters are defined for the most significant subsystems. Estimates of the significant system parameters (i. e., weight, cost) are made for those subsystems which cannot be defined in detail.

Interface requirements are established and a conceptual configuration design produced.

Figure 3-5 shows a two-view sketch of the avionics subsystem conceptual general arrangement as derived in this study.

The vehicle axes are as defined as shown here with $+Z_V$ always being maintained towards earth, thus being coincident with the local vertical and with the reference yaw axis. The reference pitch and roll axes will undergo a complete rotation about the Z axis in the vehicle $X_V - Y_V$ plane as the vehicle undergoes the necessary yaw and pitch rotation at orbital rate during transfer orbit.

The placement of components is a straightforward procedure because establishment of the axes and control maneuvers automatically position the sensing and thrusting components within certain limits. The remaining components are distributed circumferentially as uniformly as possible for mass balancing purposes and for minimizing propellant line and electrical harness lengths.

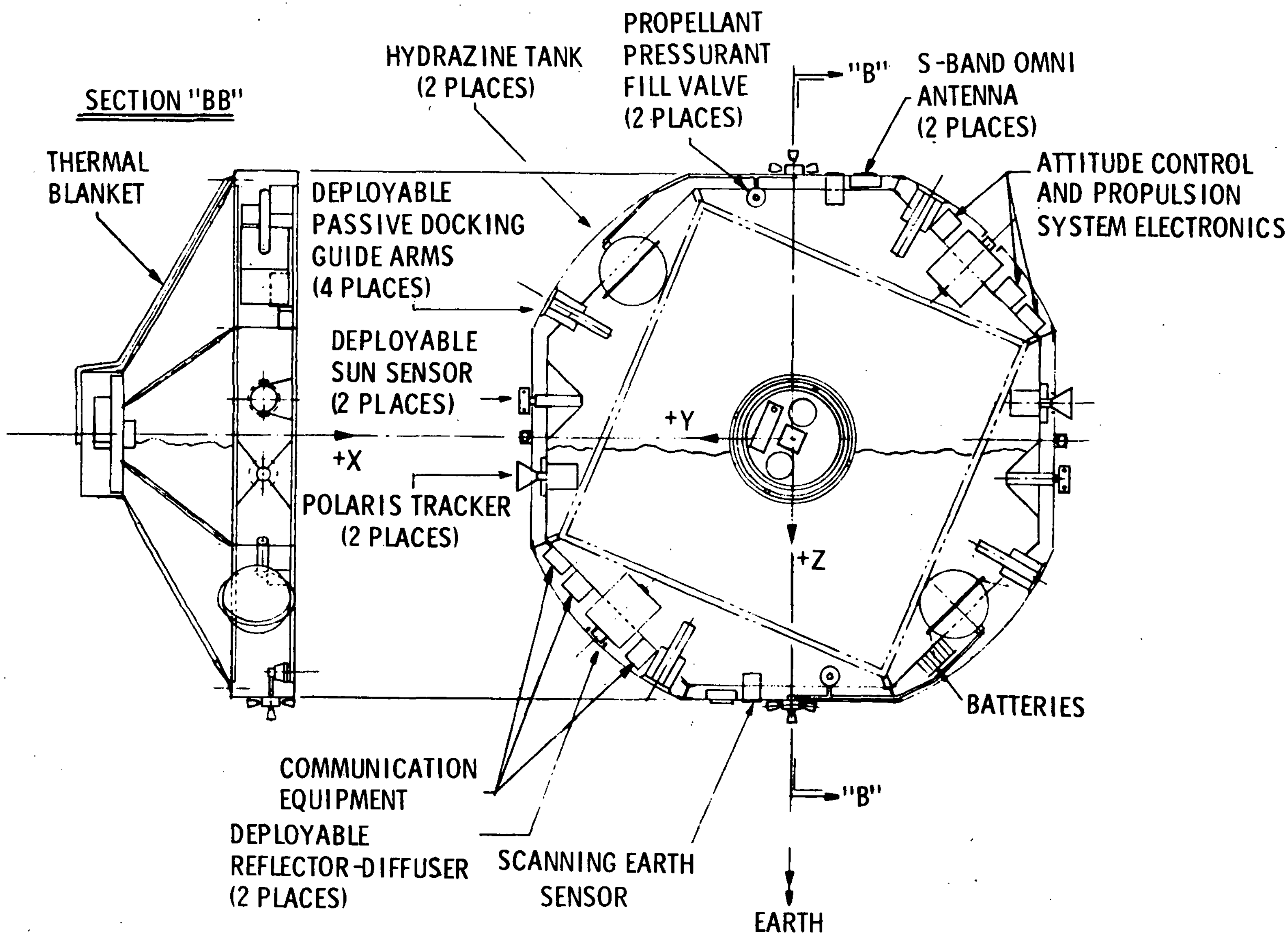


Figure 3-5. Avionics System General Arrangement

The dimension across the inner component support shell structure was selected arbitrarily to permit the square docking frame to nest inside. A large dimension means greater inner shell surface area and less packaging volume, which relates to packaging density and shell weight. A larger dimension also results in keels and bulkheads of small depth. A trade study is required to evaluate these various factors and arrive at an optimum design. As it is, there appears to be ample packaging volume in the space between the inner shell and the outer thermal surface for the avionics subsystem components.

Note the requirement for opposed sensors and antennas due to the complex maneuvering requirements.

The video/illumination/SLR platform is recessed into the power conditioning module because the required gimballing angle is inversely proportional to the distance away from the target. A trade study of the optimum longitudinal location of this platform would be necessary.

The following sections describe the avionics subsystem design requirements, the alternative design approaches considered, the selected designs, and some of the additional trade studies that would be advantageous.

3.6.1 ATTITUDE CONTROL SUBSYSTEM

The Attitude Control Subsystem (ACS) consists of the Thrust Vector Control (TVC) Subsystem, the Reaction Control Subsystem (RCS), and the sensors and trackers required to provide the vehicle attitude and thrust vector orientation functions necessary to satisfy the geocentric orbit mission requirements.

The attitude control subsystem is a six degree of freedom active control system operating in a controlled limit cycle using a combination of the ion thrusters hydrazine thrusters for control torques. The ion thrusters provide the majority of the propulsive forces during the mission operations with the hydrazine thrusters operating during periods of reactor shutdown and during the final close-in payload docking maneuvers. Attitude and rate sensors are selected by ground command and stored commands as a function of mission phase.

The control reference axis system is a set of three orthogonal axes located at the center of mass of the NEP Stage. In orbital operation, the stage is rotating at orbital rate so that the yaw (Z) axis coincides with the local vertical and is positive downward (see Figure 3-5). The yaw axis remains fixed relative to the vehicle axes in the earth reference system used here. The roll (X) axis is orthogonal to the yaw axis and lies in the orbit plane with the positive sense in the direction of the velocity vector. The pitch (Y) axis is orthogonal to both the local vertical and the orbit plane; the positive sense is to the right when looking in the direction of the velocity vector.

Table 3-5 shows the design criteria assumed for the conceptual design of the attitude control subsystem. For subsystem design purposes, the mission phases are classified as either an orbital phase or a rendezvous/docking phase, which impose two distinctly different sets of requirements on the control system.

Angular position error specifications are valid for all of the three orthogonal reference axes. Allowable angular rates are the same for any mission phase, but vary for the vehicle axes, due to the variation of inertias to be expected for the prolate-type of body configuration that the NEP Stage possesses.

As noted the translational position and rate requirements are considered negligible for orbital operations, but very critical for the rendezvous/docking operation.

The design requirements shown in Table 3-5 must be capable of being accommodated over a wide range of vehicle mass property values, dependent upon the particular payload which is being transported by the NEP Stage.

3.6.1.1 Sensors and Trackers

Figure 3-3 shows when the various ACS sensors are located on the avionics module. A horizon sensor is located with its optical axis parallel to the yaw axis. Two analog sun sensor assemblies are located around the vehicle Y axis, being elevated and shielded to avoid reflections and to prevent blockage by the stage or payload. The Polaris tracker mounting requires the

optical axis to be maintained ± 30 degrees to the reference pitch axis to accommodate orbit inclination. The trackers are mounted with axis parallel to the vehicle lateral axis (Y_V) which is orthogonal to the yaw axis. A Z axis gimbal concentric with the optical axis produces the 30 degrees motion needed. To be usable during the out of plane thrusting maneuvers, a rotating mirror is deployed in front of the sun shade to shift the optical axis a nominal 90 degrees. A three-axis gyro package is mounted so that the input axes of the gyro lie along the vehicle axes. A yaw wheel is mounted so that its axis of rotation lies along the yaw axis.

Table 3-5. Avionics Subsystem ACS/Docking Control Requirements

Position/Rate Limits		
Item \ Phase	Orbital Operation	Rendezvous Docking Operation
Angular Position (deg)	± 2	
Angular Rate (deg/sec)	Lateral Axes <0.05 Longitudinal Axis < 0.10	
Lateral Position (m)	NA	± 0.15
Translational Rates (m/sec)	NA	Lateral <0.03 Longitudinal 0.03 to 0.3

Mass Property Range

- Inertia ($kg\text{-}m^2$): 100,000 - 700,000
- Center of Mass Shift: Up to 7 Meters

- Six Degrees of Freedom
- Limit Cycle Control Law, $\theta + K\dot{\theta} = \pm n$

3.6.1.2 Thrust Vector Control

The TVC subsystem consists of the ion engines, the engine gimballing mechanisms, and the engine controls necessary to provide 3-axis control of the NEP Stage during all mission phases, except for terminal docking maneuvers and during periods when the reactor is shut down.

Table 3-6 presents the implementation scheme currently envisioned for the NEP Stage guidance, control, and propulsive functions. The three columns show the control mode, sensor utilization, and propulsive system implementation for each of the mission phases which are defined.

After separation of the NEP Stage from the Shuttle, the stage rates are sensed by a 3-axis rate gyro package, and the signals are processed by the Attitude Control Electronics to produce firing signals for the hydrazine engine of the reactor control subsystem (see Section 3.6.1.3). After the rates have been reduced by the torques of the hydrazine engines, pitch and roll axis control is switched to the Earth Sensor. After earth acquisition, yaw control is switched to the analog sun sensor. Yaw is controlled to the sun line until the star Polaris can be acquired by the gimbaled Polaris tracker. After Polaris acquisition, the reactor startup can be initiated. Once the ion engines are activated control torques are provided by the ion engines.

During the initial phase of the transfer orbit, only translational thrust is applied to the NEP Stage. When an altitude is reached at which out-of-plane thrusting for orbit inclination change becomes effective, yaw control is transferred to the analog sun sensor array. The stage is rotated at orbital rate about the yaw axis to produce alternate translational and normal thrusting by the ion engines. A reaction wheel controls the yaw slew wheel and the analog sun sensor assembly is determined by a stored program. Ground updates are required for orbital changes. The Polaris tracker updates the orbit position data twice an orbit. The yaw program controls the yaw slew wheel rotational speed as a function of orbit altitude. The wheel direction is reversed at the orbit nodes. The yaw program adds a bias to the yaw analog sensor as a function of orbit position to provide an attitude loop around the yaw loop. To maintain yaw control throughout the orbit, control is switched from one set of sensors to another. During eclipse periods, yaw control is switched to the yaw gyro. An appropriate bias is generated to continue the yaw program.

Table 3-6. Avionics Subsystem ACS/Docking Control/Propulsion Implementation

Mission Phase	Control Mode	Sensors	Impulse
Initial Stabilization			
Separation Acquisition	Rate Limiting Sun/Earth Acquisition and Hold	3-Axis Gyro Package Sun Sensor, Horizon Scanner	Auxiliary Propulsion
Transfer Orbit (Deployment and Operational)			
Normal	Earth/Sun Hold	Horizon Scanner, Sun Sensor	Ion Engine System
Eclipse	Earth/Rate Control	Horizon Scanner, 3-Axis Gyro Package	
Gyro Update	Earth/Star Hold	Horizon Scanner, Polaris Tracker	
Synchronous Orbit Operation			
Closing Orbit	Earth/Star Hold	Ground Tracking Horizon/Sun Sensors	Ion Engine System
Rendezvous Docking	Gyro Hold Gyro Hold	SLR Video, 3-Axis Gyro Package	Auxiliary Propulsion

After the orbit altitude and inclination changes have been completed, the attitude control subsystem is commanded to the horizon sensor-Polaris tracker configuration. The subsystem remains in that configuration until the rendezvous and docking phase.

The required NEP Stage attitude for the period of the rendezvous maneuver is computed on the ground using orbital element information of both the NEP Stage and the spacecraft that the stage is to rendezvous and dock with. Control of the NEP Stage is transferred to the 3-axis gyro package operating in the attitude mode. The stage is slewed to the required attitude using the ion thrusters for control and slew torques. Once the required attitude is reached, the ion thrusters provide the ΔV needed to reach docking range. Docking range is reached when either the Scanning Laser Radar (SCR) or the video subsystem observes the target spacecraft. Upon reaching the docking range, the ion thrusters are throttled and the hydrazine thrusters are used to provide the required control torques.

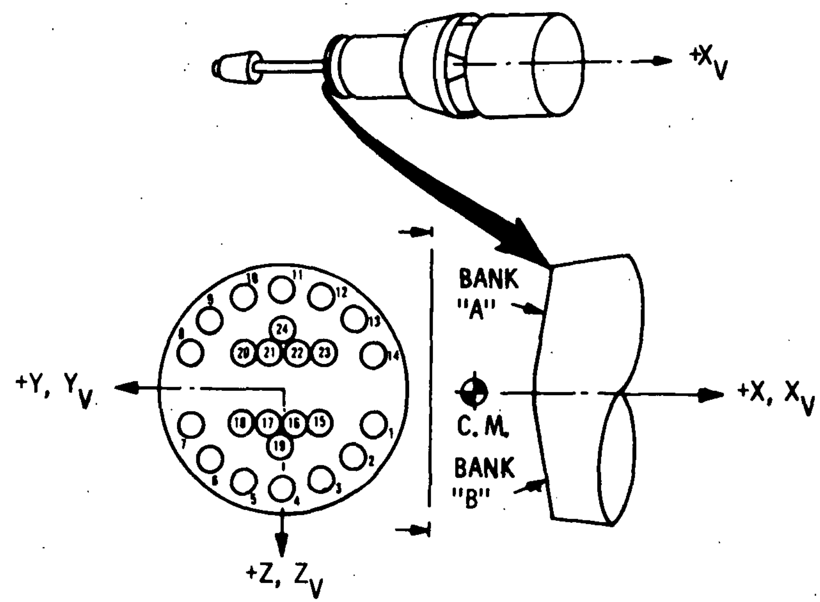
Docking maneuvers are accomplished by ground control using SLR and video data. The final docking maneuvers are done using video data. The gyros are now operated in the rate mode.

Figure 3-6 shows the conceptual approach to the thruster actuation sequencing that is required for utilization of the electric ion engine thrust subsystem in the attitude control and thrust vector control mode.

This sequence assumes 2-axis gimballing capability for the outer row of engines (Numbers 1 through 14), and individual throttling control capability for all of the thrusters.

3.6.1.3 Reaction Control Subsystem

It is assumed that a Reaction Control Subsystem (RVS) using hydrazine thrusters would be designed as an integral part of the control system. Figure 3-7 presents the design considerations made with respect to this subsystem.



NOTE:

1. LATERAL TRANSLATION FORCES PRODUCE ROTATIONS ABOUT Y AND Z AXES UNLESS C. M. IS IN THRUST PLANE.
2. FOR C. M. LOCATIONS FORWARD OF ENGINE ARRAY PLANE, TRANSLATIONAL FORCE SEQUENCES CHANGE SIGNS.

ROTATIONAL TORQUES	TRANSLATIONAL FORCES
+X: TANGENTIAL GIMBAL	+XX: GROUPS SYMMETRICAL ABOUT AXIS
-X: TANGENTIAL GIMBAL	-XX: NOT POSSIBLE (MUST ROTATE VEHICLE 180°)
+Y: THROTTLE BANK "B"	+YY: GIMBAL IN PLANE ∥ Y AXIS
-Y: THROTTLE BANK "A"	-YY: GIMBAL IN PLANE ∥ Y AXIS
+Z: THROTTLE 12-14, 1-3	+ZZ: THROTTLE BANK "B"
-Z: THROTTLE 5-10	-ZZ: THROTTLE BANK "A"

Figure 3-6. NEP Stage Ion Thruster Actuation Sequences

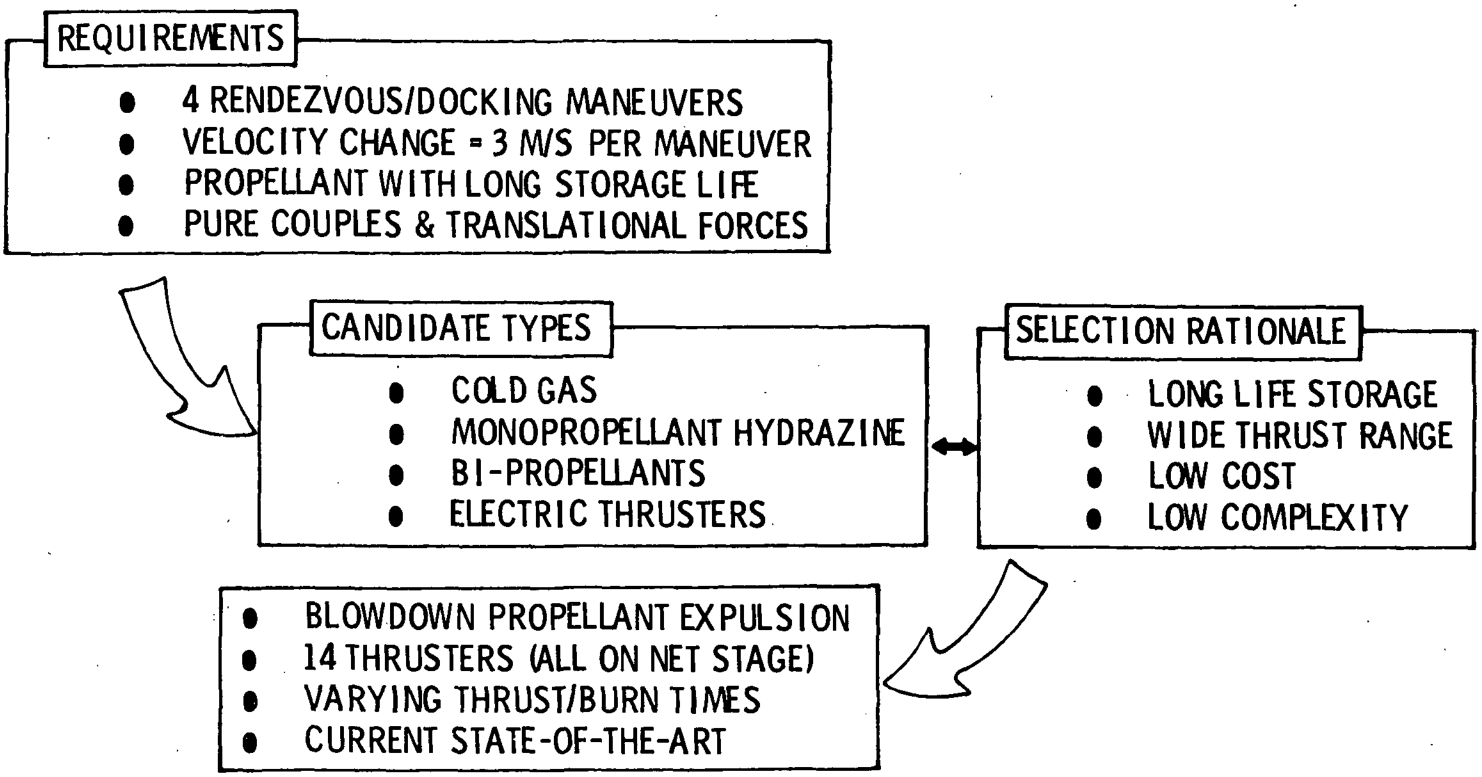


Figure 3-7. Avionics Subsystem RCS

Impulse requirements for the hydrazine thruster system have been arbitrarily established at a conservative level of 3 m/sec per docking maneuver. The ion engine and SLR system should be able to get the final rendezvous and docking ΔV requirement to a lower level than 3 m/sec since the final rendezvous maneuver can start at > 300 km distance. The impulse corrections can be made by the ion engines acting in an autonomous mode with the SLR system down to the point where the video/ground control loop would take over. The remaining velocity error at this point is limited only by the accuracy of the SLR sensing system and the associated TVC, and could be very small.

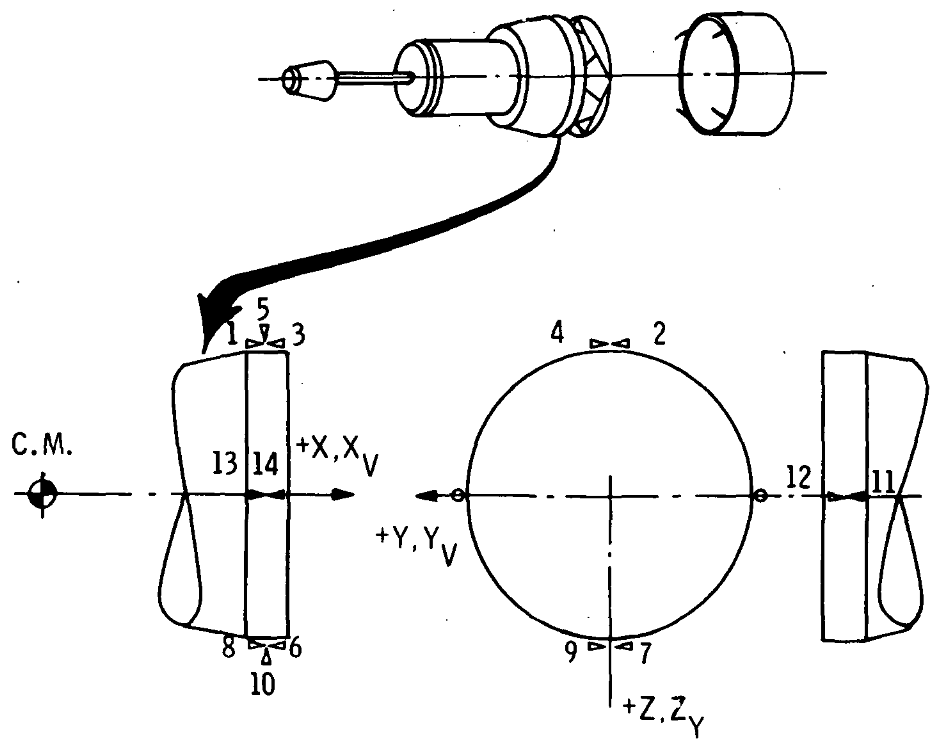
Candidate types considered for auxiliary propulsion usually include many different types from cold gas to electric thrusters, with final selection being made on the specific cost versus mission performance requirements.

Current spacecraft design criteria for long life missions and good performance are developing a trend to selection of monopropellant hydrazine systems. These systems are selected over cold gas systems on a weight performance basis and over bi-propellants on reliability of operation throughout a long mission lifetime. The trade between a hydrazine system and an electric thruster would be made on the basis of thrust requirements, reliability, and cost. At this point in time, the thrust requirements for this NEP system are considered loose and the hydrazine system is selected on the basis of allowing greater design flexibility with respect to control torques and forces.

The selected RCS is a blowdown monopropellant hydrazine system consisting of two tanks, fill and drain valves for propellant and pressurant that must incorporate a remote control or automatic "in-flight refueling" capability for interfacing with the Propellant Logistics Depot (PLD), filters, pressure transducers, temperature sensors, and 14 engines of three different thrust levels with redundant control valves. The thrust levels are selected for compatibility of operation in a pure translational or pure rotational mode. The tanks are 49 cm in diameter to hold the 73 kg of propellant plus the necessary pressurant which is required to provide the 40,000 lb-sec of impulse per mission (based upon a specific impulse of 250 sec). Figure 3-5 indicates the location of the thrusters, which are mounted in four clusters of engines.

Figure 3-8 presents the actuation sequence assumed for the reaction control subsystem to provide the decoupled angular rotation torques and net lateral forces required for the sensitive maneuvers required during active docking with a payload or the PLD.

As conceived in this study, the RCS hydrazine thrusters are all located on the avionics subsystem itself; whereas, it may be advantageous in the final design to locate thruster assemblies at the maximum dimension allowable to increase available torque or to decrease fuel consumption. Trade studies will be required to make the final selection.



ROTATIONAL TORQUES	TRANSLATIONAL FORCES
+X: 2/9	+XX: 1/8, 11/13
-X: 4/7	-XX: 3/6, 12/14
+Y: 3/8	+YY: 2/7 + 12/13
-Y: 1/6	-YY: 4/9 + 11/14
+Z: 11/14	+ZZ: 5 + 3/8
-Z: 12/13	-ZZ: 10 + 1/6

NOTES:

1. FOR C.M. LOCATIONS FORWARD OF NET STAGE THE COUNTERACTING COUPLE FORCES INTERCHANGE FOR THE ± LATERAL TRANSLATION FORCES.
2. FOR MAINTENANCE OF PURE TRANSLATION FORCES VARIABLE THRUST LEVELS AND/OR BURN TIMES ARE REQUIRED AS FUNCTION OF THE C.M. LOCATION.

Figure 3-8. RCS Hydrazine Thruster Actuation Sequence

For this concept the lateral thrusting will very likely produce rotational torques about the vehicle center-of-mass (CM) which will be counterbalanced by simultaneous thrusting of the angular torque thrusters. Thruster levels will be selected to be compatible with the allowable position errors and rates. For example, if we consider a unit force of 1 Newton (0.22 lb) at the maximum moment arm of 4.6 m, it would take 90 seconds for an angular change of 2 degrees for a vehicle with $I = 5 \times 10^5$ kg-m^2. If the initial allowable rate was 0.1°/sec, in 90 sec, the vehicle would rotate 9 degrees and consequently the available torque is insufficient to accommodate a position error of less than 7 degrees in this case. A detailed error analysis is necessary for the next level of design.

It would appear to be advantageous from an overall mission and design standpoint if the primary electric ion propulsion subsystem could be used for all propulsive functions, because the addition of a separate propulsion subsystem adds cost and weight, and may decrease reliability. It also appears to be feasible to accomplish all maneuvers with the ion engine by itself if:

1. The resultant thrust vector of the engine array goes through the vehicle CM (as in the reference end thrust NEP Stage configuration) and control torques are available with the individual engine thrust vectors.

2. The capability to throttle the engines is sufficient to provide the range of velocities and accelerations that certain maneuvers (particularly docking) may require.

3. The reactor/ion engine system can indeed be operating in all maneuver modes.

Condition 1 above is automatically met for the longitudinal thrust configurations but would present a severe, but probably not insurmountable, problem for a side thrust configuration. The ring of engines concentric to the CM provides torque capability about lateral axes by variable throttling. Roll and yaw control are obtained by engine gimballing (can also be accomplished by fixed cant).

For the mission modes in which the NEP Stage is performing rendezvous with unmanned systems, Condition 3 above can probably be met, allowing for an initial uncontrolled separation mode from the Shuttle.

Condition 2 then seems to present the only constraint upon usage of the ion engine system by itself for all maneuvers, and this appears to be a constraint only for the active docking maneuver, in which delicate control of the vehicle rates is required. The main drawback of the ion engine by itself would be the inability of the engines to reverse thrust and slow the vehicle down while maintaining the required vehicle attitude for rendezvous and docking. If the basic engines can be throttled down to very low levels (5 percent of maximum thrust or less), then it would seem feasible to provide a minimum number of opposing ion thrusters (e. g., three) to provide the reverse thrust that may be needed. Since, however, the hydrazine system is needed before startup and possible inadvertent shutdown, a mixed mode operation is probably optimum for Condition 2.

Consideration of the aforementioned facts indicates that additional analysis is needed to determine the possibility and optimality of operation of the NEP Stage with adaptive control. This would involve determination of throttling capacity, thrust build-up time from a standby mode, control forces versus acceleration required, complexity of operating different thrusters or groups of thrusters at variable levels, optimum number of ion engines, etc.

3.6.2 FLIGHT COMMAND SUBSYSTEM

The Flight Command Subsystem (FCS) is comprised of the Central Computer and Sequencer (CC&S) and the Flight Data Subsystem (FDS).

3.6.2.1 Central Computer and Sequencer (CC&S)

The primary function of the central computer and sequencer is to maintain control of the NEP Stage, both thrust vector control and reactor control. Typical requirements of the CC&S include providing comments to the TVC subsystems, commanding reactor startup, control of the reactor after startup until reactor is capable of supplying power (i. e., reactor is up to 30 percent power), commending switch over to reactor control, reactor drum stepping control, and cesium temperature control.

Key components that comprise the CC&S and their respective masses are shown in Table 3-7.

Table 3-7. Key Central Computer and Sequencer Components

Component	Mass (kg)
Control Computer	80
Timing Synchronizer	4
Control and Conditioning Logic	20
Power Conditioning	15

3.6.2.2 Flight Data Subsystem (FDS)

The primary function of the flight data subsystem is to monitor the operational status of the NEP Stage. Typical monitoring requirements of the FDS are:

1. Docking and rendezvous approach
2. Voltage from the thermionic fuel elements (TFE's)
3. Current from TFE
4. Neutron level from reactor
5. Average neutron level
6. Coolant temperature
7. Coolant pressure
8. Control drum position
9. Reactor cesium temperature
10. Contact closure

Key components that make up the flight data subsystem are listed in Table 3-8.

Table 3-8. Key Flight Data Subsystem Components

Component	Mass (kg)
Approach Guidance	15
Measurement Processer	6
Data Storage	40
Power Conditioning	20

3.6.3 FLIGHT TELEMETRY SUBSYSTEM

The requirements and implementation approach established for the conceptual design of the Flight Telemetry Subsystem (FTS) are shown in Figure 3-9. The FTS provides the RF link(s) for four different functions (partially interrelated):

1. Telemetry. The telemetry subsystem receives component diagnostic information as analog or digital signals which it transmits to the ground via an RF link at the operating frequency.

2. Tracking. The tracking components will provide the information required for orbit determination during all orbit maneuvers other than rendezvous and docking. A range and range rate (RARR) transponder operating at S-band will receive and transmit the necessary tracking signals.

3. Command. The command subsystem receives the necessary functional commands for NEP Stage operation via an RF link from the ground control station(s).

4. Rendezvous and Docking. The rendezvous and docking components provide the two-way RF link required during closing and docking maneuvers. This includes transmission of the video information to ground control and reception of the responding ground commands. An autonomous scanning laser radar (SLR) is used for initial closing maneuvers from radar pick-up to the point where the the video picture resolution is sufficient to permit switching to ground command.

The block diagram for the flight telemetry subsystem is shown in Figure 3-10.

The telemetry capacity assumed is the 1 KBPS value used as standard for the MSFN system. It is assumed that a 26 meter diameter ground antenna would be within view from any orbital longitude and inclination for the time period of this mission.

The implementation is essentially defined with current state-of-the-art components, using a 2 W solid state output amplifier for TT&C and a 100 W TWT (or perhaps solid state) amplifier for the video carrier. As noted before the video picture quality could perhaps be reduced by 10 dB in which case the 100 W transmission could be reduced to 10 W.

REQUIREMENTS

- ORBITAL OPERATIONS
  - PERIODIC TELEMETRY TRANSMISSION
  - PERIODIC TRACKING TRANSMISSION
  - PERIODIC COMMAND RECEPTION
- RENDEZVOUS/DOCKING
  - 2-WAY COMMUNICATION WITH GROUND
    - SLR RANGE/RANGE RATE
    - VIDEO PICTURE
    - COMMAND
  - RANDOM VEHICLE ORIENTATION AND ORBITAL POSITION
- COMPATIBLE WITH MSFN AND DSN
- 26 METER DIAMETER EARTH TERMINAL ANTENNA

IMPLEMENTATION

- OMNI-DIRECTIONAL COVERAGE
  - 2 OPPOSED ANTENNAS (RF SWITCH)
  - CIRCULAR POLARIZATION
- RF POWER
  - 100 W FOR VIDEO CARRIER
  - 2 W FOR TELEMETRY AND TRACKING
- CURRENT STATE-OF-THE-ART

Figure 3-9. Flight Telemetry Subsystem

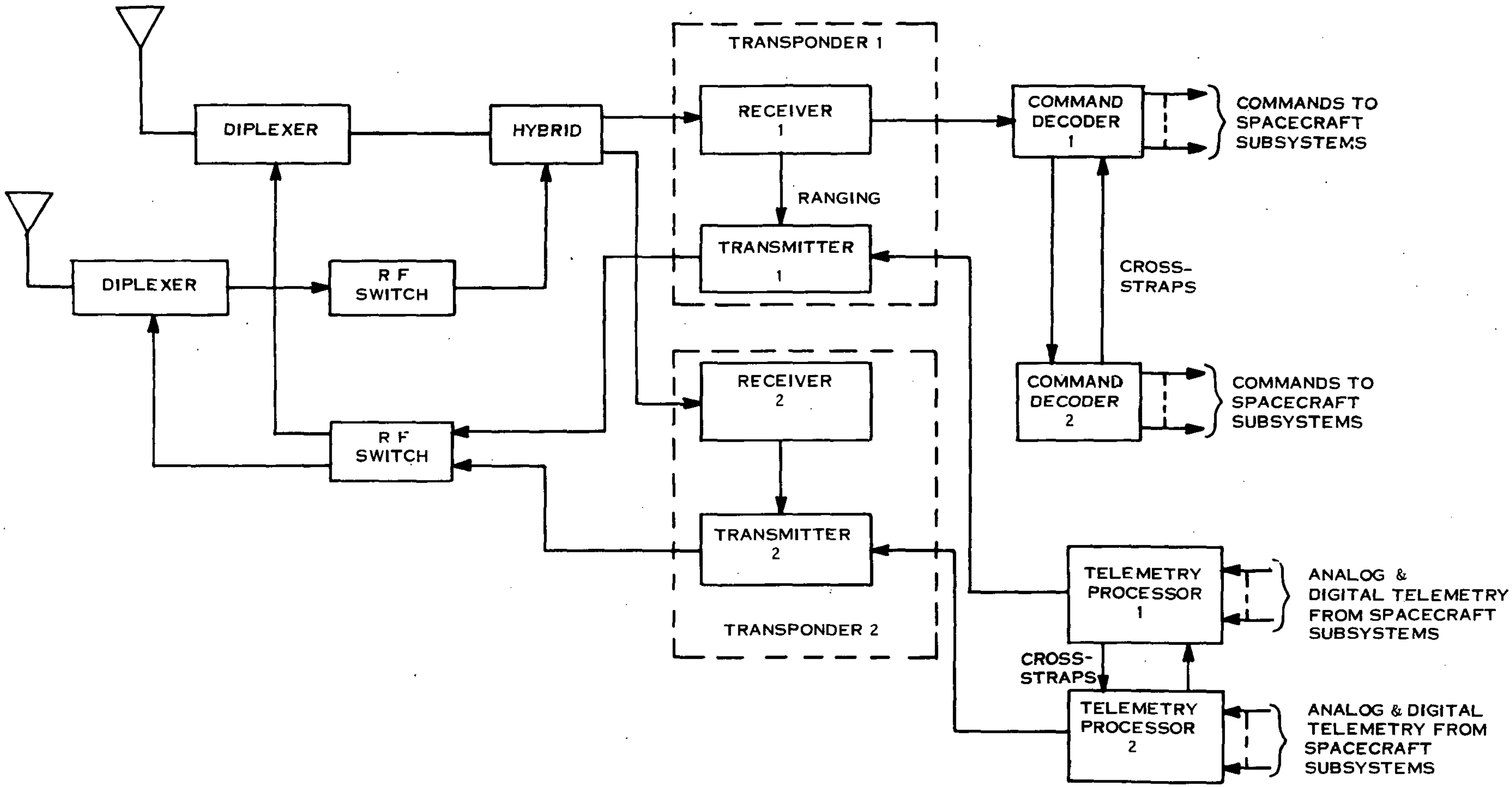


Figure 3-10. Flight Telemetry Subsystem Block Diagram

The antenna design requirement is to provide a circularly polarized omnidirectional coverage antenna pattern for the communications carriers operating at a receive frequency of 2.2 GHz and a transmitting frequency of 1.7 GHz.

The antenna concept selected to meet these design requirements is a cavity-backed spiral designed for an operating center frequency of ≈ 2 GHz which will provide the necessary bandwidth for both reception and transmission of the carriers as required. Two antennas are placed opposite each other on the outer surface of the avionics subsystem, as shown in Figure 3-5, for provision of the desired omnidirectional coverage.

The most severe performance requirement for the communications link is transmission of the video signal to the ground control station. Link analysis of a previous study (Reference 3-5) used an RF bandwidth of 10 MHz and an EIRP of 40 dBW for an FM TV signal operating at FM threshold (C/N ≅ 10 dB after losses and margin) for the video link of a remote manipulator servicing a satellite in geosynchronous orbit. If the same operating parameters are selected, and an omni-antenna (which provides an EIRP ≈10 dBW with 10 W of RF power) on the NEP Stage and 6 dB for losses and margin are assumed, then the receiving ground terminal would require a G/T value of ≈ 39 dB $^{\circ}K^{-1}$. If operation is assumed to be with a 26 m diameter dish (which is the current plan for the MSFC Chemical Tug - Reference 3-6), the receiving gain would be about 51 dB (at f = 1.7 GHz), requiring the receiving system noise temperature to be less than 16°K (12 dB) for the parameters previously specified. This low temperature is not practicable and consequently, the RF power output from the NEP Stage would have to be increased or the RF bandwidth decreased. The addition of 10 dB to the power output (to 100 W) permits the noise temperature to increase to 160°K which is satisfactory. A value of 33 percent efficiency using TWT output power amplifier(s) is conservative for current state-of-the-art projections and is used for this NEP concept.

The interplanetary missions planned for the NEP Stage require the transmission of large quantities of data over distances up to several billion kilometers. For a particular mission, the quantity of data which can be transmitted for a time-period is directly related to the

transmitter radiated power and the data-coding efficiency. Therefore, arriving on target-rendezvous with the thermionic reactor electrical capacity no longer necessary for ion propulsion, the NEP Stage has a large communication capability using the available power.

The amount of radiated power necessary for communications is dependent upon receiving a signal with a minimum strength above the background noise to ensure understanding the information. The type of data -coding dictates this minimum signal-to-noise ratio (SNR). Pioneer-65, for instance, used phase-shift keying (PSK) requiring approximately 8 dB minimum SNR. Mariner-69 used block encoded biorthogonal comma-free code with a resulting improvement in SNR required, of approximately 2.2 dB over PSK. An additional improvement in SNR of approximately 3.4 dB is achieved in Pioneer-10 by using convolutional encoding in the space probe and sequential decoding at the receiver (Reference 3-7). Consequently, systems using convolutional encoding require approximately 1/4 the transmitted power of the conventional PSK system. The major disadvantages of convolutional encoding is the amount of computer equipment necessary for decoding and decoding time-delay.

The NEP Stage, with an arbitrarily assumed 4.6 m diameter parabolic communication antenna, is capable of transmitting video at 10^6 bit/second rate from Jupiter, expending 16 kilowatts of electrical power for PSK coding and 4 kilowatts with convolutional encoding. High quality color video requires a data rate of 10^7 bits/second, which from Jupiter requires 40 kilowatts of power using convolutional encoding. The multiplier for radiated power when using other than the 15-foot antenna are shown in Figure 3-11.

Results of calculations showing the effect of transmitted power upon data rate for the 4.6 m dish are shown in Tables 3-9 and 3-10 for uncoded PSK and convolutional encoding, respectively. The following assumptions entered the calculations:

Frequency	3.2 GHz (S-band)
NEP Stage Antenna	4.6 meter parabolic dish, 39 dB gain
Ground Antenna	64 meter deep space instrumentation facility (DSIF), 61.4 dB gain
Noise Temperature (DSIF)	30°K
Signal Margin	6 dB

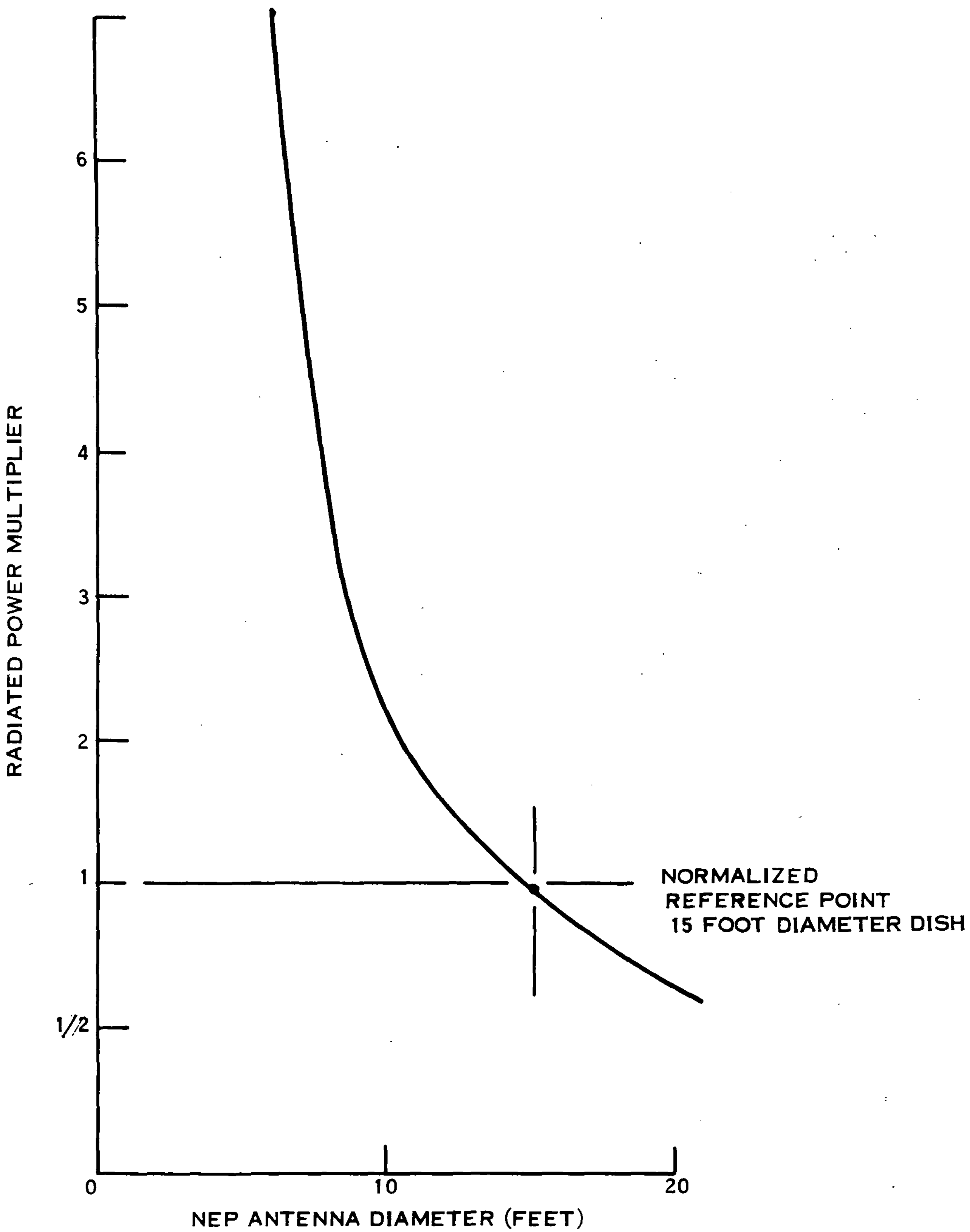


Figure 3-11. Antenna Size Effect Upon Radiated Power

Table 3-9. Transmitted Power Requirements Uncoded PSK

Mission Objective	Bit Rate 10^3 B/S	10^4 B/S	10^{6*} B/S
	watts	watts	watts
Mercury	0.4	4	400
Asteroid (Ceres)	1.6	16	1,600
Jupiter	16.0	160	16,000
Saturn	22.0	220	22,000
Uranus	80.0	800	80,000
Neptune	192.0	1,920	192,000
Pluto	320.0	3,200	320,000
Sun Probe	664.0	6,640	664,000

*Necessary for Real-Time television or radar data transmission.

Table 3-10. Transmitted Power Requirements
Convolutional Encoding

Mission Objective	Bit Rate 10^3 B/S	10^4 B/S	10^{6*} B/S
	watts	watts	watts
Mercury	0.1	1	100
Asteroid (Ceres)	0.4	4	400
Jupiter	4.0	40	4,000
Saturn	5.5	55	5,500
Uranus	20.0	200	20,000
Neptune	48.0	430	48,000
Pluto	80.0	800	80,000
Sun Probe	166.0	1,660	160,000

*Necessary for Real-Time television or radar data transmission.

For comparison, Pioneer 10 has an 8-watt transmitter, a 2.75 m diameter dish and uses convolutional encoding. The spacecraft is designed to transmit at Jupiter range with a rate of 10^3 bits/second.

3.6.4 VIDEO/LIGHTING SUBSYSTEM

Figure 3-12 presents the design requirements assumed for the video/lighting subsystem and the resulting implementation concept.

The video picture quality requirements are assumed to be considerably degraded from normal television standards, but the lower resolution, slow-scan approach is considered adequate for monitoring the simple docking function. S/N output is assumed high at this point and could perhaps be reduced by as much as 10 dB with associated RF power output reduction.

The illumination range requirement is considered to cover any situation, although it would be possible to choose to dock during that period of the orbit which would give most favorable illumination from the sun.

The selected video/lighting subsystem resulting from these requirements is an Apollo-type TV zoom camera mounted with associated controlled light source affixed to the platform with the same orientation as the camera or with a moveable reflector/diffuser. A schematic of the video/lighting subsystem is shown in Figure 3-13.

The high resolution TV camera is focused by ground command; pan and tilt requirements are effected by the moveable platform upon ground command.

Illumination will be provided by incandescent lamp(s) if the target side is totally on the dark side, or by use of a moveable reflector/diffuser if the specific target to be viewed is in the shadows on a side being brightly illuminated by the sun.

REQUIREMENTS

- VIDEO PICTURE
  - 250 TO 500 LINES PER INCH
  - >10 FRAMES PER SECOND
  - S/N = 40 TO 50 dB
- ILLUMINATION
  - FULL SUN TO COMPLETE SHADOW

IMPLEMENTATION

- TV CAMERA
  - ZOOM LENS
  - 2-AXIS GIMBAL
- ILLUMINATION
  - INCANDESENT LAMP
  - REFLECTOR/DIFFUSER

Figure 3-12. Video Lighting Subsystem

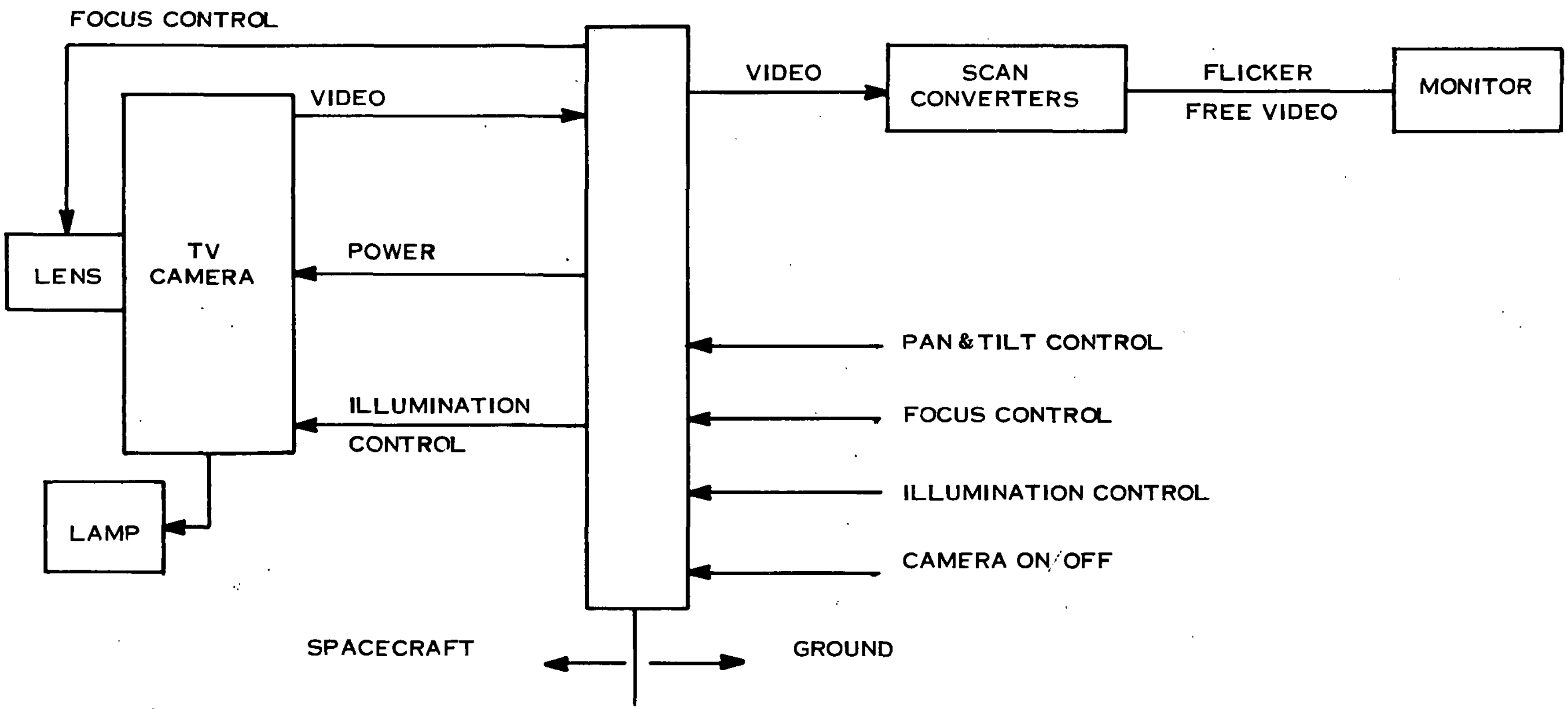


Figure 3-13. Video/Lighting Subsystem Schematic

If the target is brightly illuminated by the sun, an automatic exposure control system in the TV camera will maintain constant average illumination of the pick-up tube photocathode.

3.6.5 DOCKING SUBSYSTEM

The docking subsystem design must meet the subsystem design requirements as defined in Table 3-11. Three docking subsystem designs were investigated. The following sections delineate these docking concepts.

Table 3-11. Docking Control Specifications

Baseline Configuration	- Miss Distance	± 0.15 m
	- Miss Angle (each axis)	± 1 deg
	- Longitudinal Velocity Control	0.03 to 0.3 m/sec
	- Lateral Velocity Control	0 to 0.03 m/sec
	- Angular Velocity	± 0.1 deg/sec
Spinning Target	Same as above, Plus:	
	Spin Speed Control	± 0.1 deg/sec relative to the target spin rate

3.6.5.1 Cooperative Three-Axis Docking Concept

The most straightforward requirements and subsequently the one used to define the baseline docking subsystem is that which requires mating with a cooperative three-axis spacecraft that has been designed for the purpose. The assumption is that the spacecraft will be in a stable orientation mode as the NEP Stage performs the rendezvous and docking maneuver being guided by ground control. Consequently, no manipulator arms or extendable-boom video cameras are assumed for this concept.

The design approach, illustrated in Figure 3-14, is that defined for the baseline MSFC Chemical Tug (Reference 3-8), and for this conceptual design, the same dimensions are retained. The design consists of a square frame attached to the avionics subsystem by eight actuators for extension/retraction and energy absorption.

The docking mechanism incorporates deployable/retractable structures that permit adaptation to all standardized docks that would be of interest. This could be as simple as smaller frame(s) deployed and retracted in a manner similar to the basic docking frame. Conversion of this dock to a passive dock is readily achieved by retraction of the active frame assembly so that a mating dock would engage the four passive guiding arms as shown in the figure.

The video/lamp SLR components are assumed mounted on a separate platform on a two-axis gimbal to permit view of any specific location on the target. Separate pointable reflector/diffusers are shown mounted at the vehicle extremities in order to reflect sunlight as needed for illumination.

The passive side of this docking assembly consists simply of four passive-guiding arms with provision for acceptance of a latching assembly from the active square frame. After contact and latching are effected, the actuators are retracted to a locked position for the orbital thrust operations. Thrust force and disturbance torques of the NEP Stage are essentially negligible for design purposes so the docking assembly can be an extremely lightweight design.

The latching mechanism to secure the mated vehicles consists of spring-motor driven pins or gears on the active dock frame to engage with detents or mating gears on the passive lugs. After engagement, the passive lugs are snubbed tightly against compression spring bumpers on the frame by the motor drive which is then locked in the secured position. Release and separation of the mated vehicles is accomplished by reversal of the above procedure, utilizing the stored energy in the spring to effect the separation.

The weight estimated of the baseline docking subsystem is presented below:

Item	Mass (kg)
Dock Structure	15
Actuators	14
Latches	6
	35

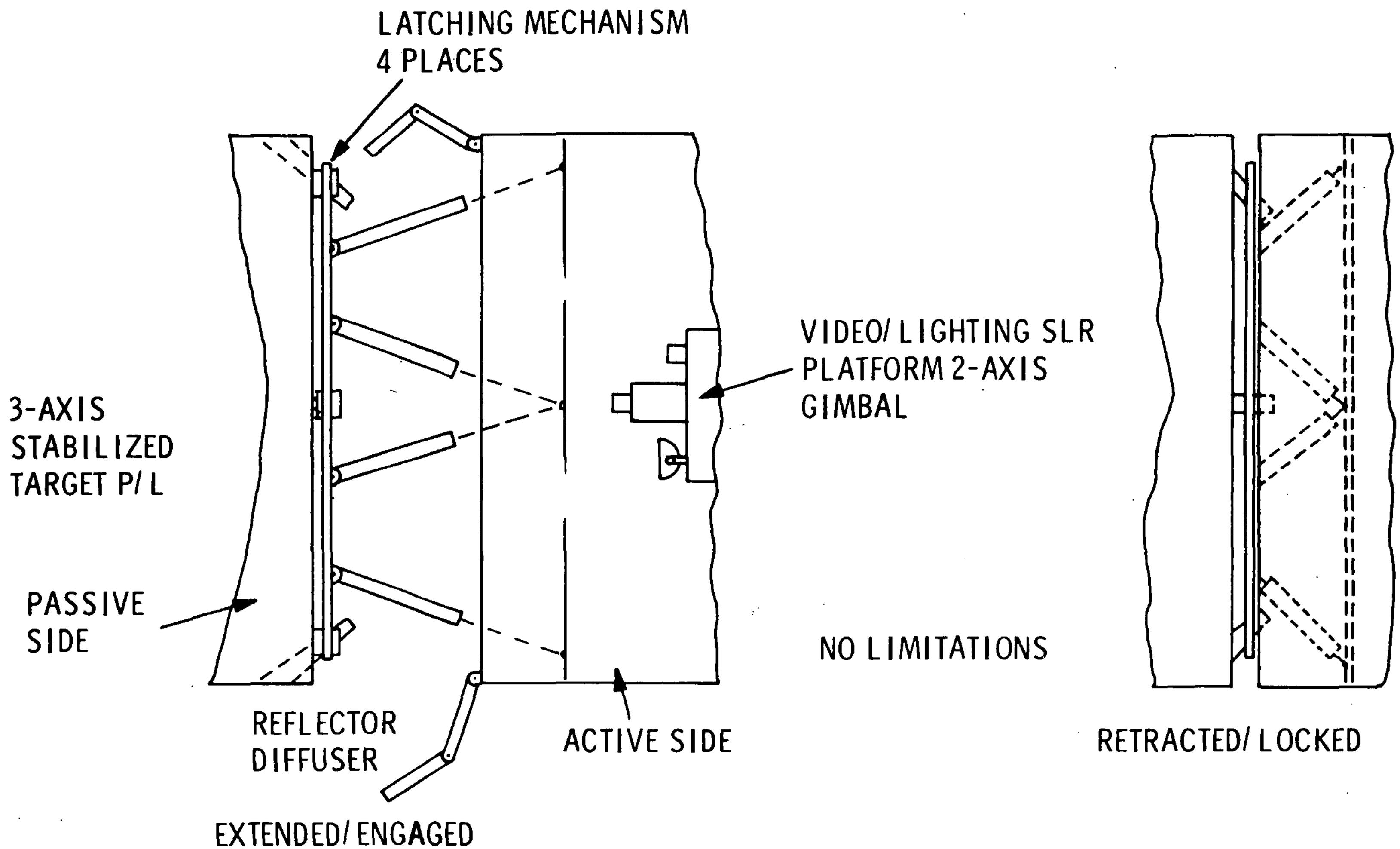


Figure 3-14. Avionics Subsystem Docking Assembly Showing Cooperative Three-Axis Docking Concept

3.6.5.2 Docking Concept for Spin-Stabilized Payloads

A second docking concept is to provide mating with a spinning target which has its spin axis coincident with the docking axis. This situation might occur if a spin-stabilized spacecraft would experience a bearing seizure between the spinning and despun portions of the vehicle and would be able to maintain a small mutation angle about this nominal spin axis. This would create a situation where it would be necessary to either despin the target or spin up the NEP Stage docking assembly to the target spin rate. It would be unrealistic to assume full despin and stabilization capability on a malfunctioning target vehicle, so provision is provided to spin the docking mechanism.

This docking concept is depicted in Figure 3-15. This concept assumes the design of a passive lug system in the target spacecraft which would be concentric with the spacecraft spin axis. The docking subsystem is basically the same as the previous assembly described, but with the additional capability to spin the docking frame and video/lighting platform assemblies. This will permit matching of the spin rate of the target payload and, as long as coning of the target is within allowable small limits, the docking can be accomplished in the same manner as the previous concept. After attachment the entire assembly of payload and docking frame would be despun.

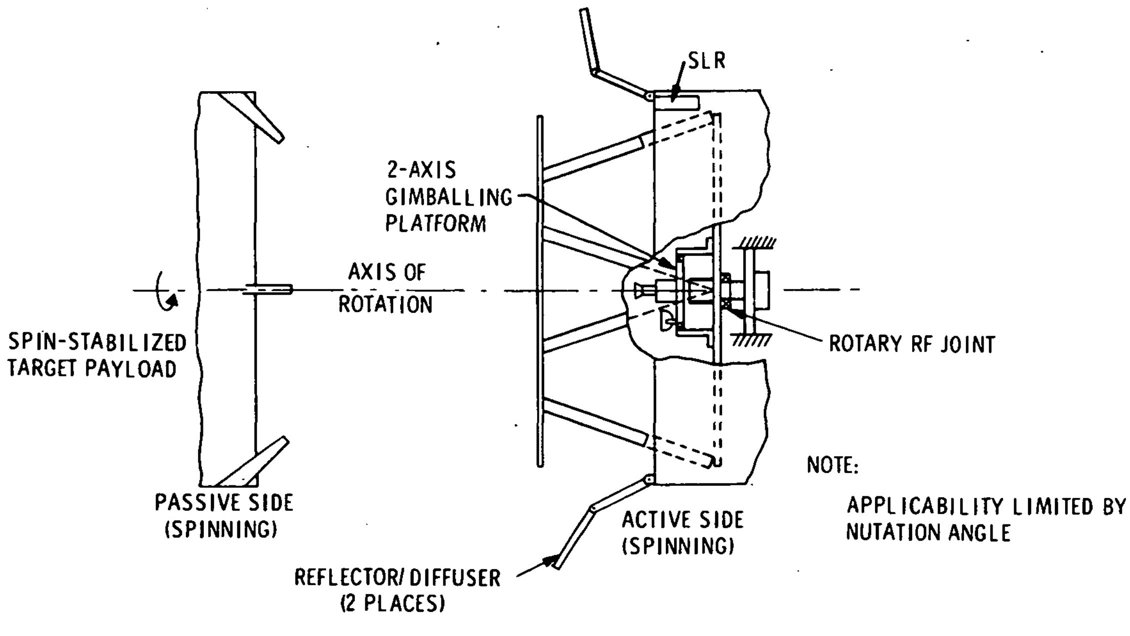


Figure 3-15. Avionics Subsystem Docking Assembly Showing Docking Concept for Spin-Stabilized Loads

The video camera(s) are mounted on the spinning dock assembly in order to achieve a "despun" picture. This will require transmission of the signal(s) across a rotating joint via an RF link or some other means. The most difficult design problem of this concept will likely be the transmission of the video signal across the rotating joint. For a single camera on the dock, an RF rotary joint using concentric waveguide operating at some frequency > 2 GHz would be suitable and is well within current state of the art. Use of a light emitting diode (LED) to modulate the video signal across the joint to a photosensitive detector on the other side for video circuitry pick-up and transmission to the ground appears to be a second feasible alternative. A requirement for multiple cameras will complicate the transmission across the rotary joint and will require RF or optical multiplexing of the multiple signals prior to transmission across the joint. If bandwidth requirements permit, the most desirable approach would be to modulate the multiple signals as subcarriers on a single RF carrier for transmission across the joint as above

3.6.5.3 Docking Concept for Randomly Tumbling Payloads

The docking concept shown in Figure 3-16 is one concept that might be considered for docking with a payload which is randomly tumbling with no chance of alignment to a passive lug docking assembly as discussed previously. Here it would be assumed that some sort of attachment device would exist on the external surface of the payload and that manipulator arms would be added to the NEP dock to implement the attachment.

The most likely situation that would exist for an uncooperative target would be for the target to be spinning and nutating about some axis not concentric with the passive docking assembly of the target. For this situation, all of the docking assembly, except for the reflector/diffuser would be spun at the target payload spin rate. The spinning active docking assembly of the NEP Stage would be aligned with the spin axis of the target, and manipulator arms would be extended to attach to the target at surface attachment points. Closing would be accomplished as in the other concepts and final attachment made by the video/ground control loop.

After payload attachment, the assembly could be despun, retracted, and locked; or else the payload could be released for a normal docking if a stable orientation/mode could be achieved.

This latter requirement may require the existence of an emergency momentum wheel in the target which would be spun up about an axis parallel to the docking axis by command from the NEP Stage.

This concept would require the addition of manipulator arms and additional video/lighting capacity to the basic docking concept.

3.6.6 THERMAL CONTROL SUBSYSTEM

Figure 3-17 presents the preliminary requirements and implementation concept derived for the thermal control subsystems.

The range of thermal dissipation requirement is wide; i.e., from potentially long periods with limited available power, and consequently minimal dissipation, during reactor shutdown, to the condition of maximum dissipation of around 500 W with unlimited power available during reactor operation. Solar insulation may vary widely throughout the mission as well.

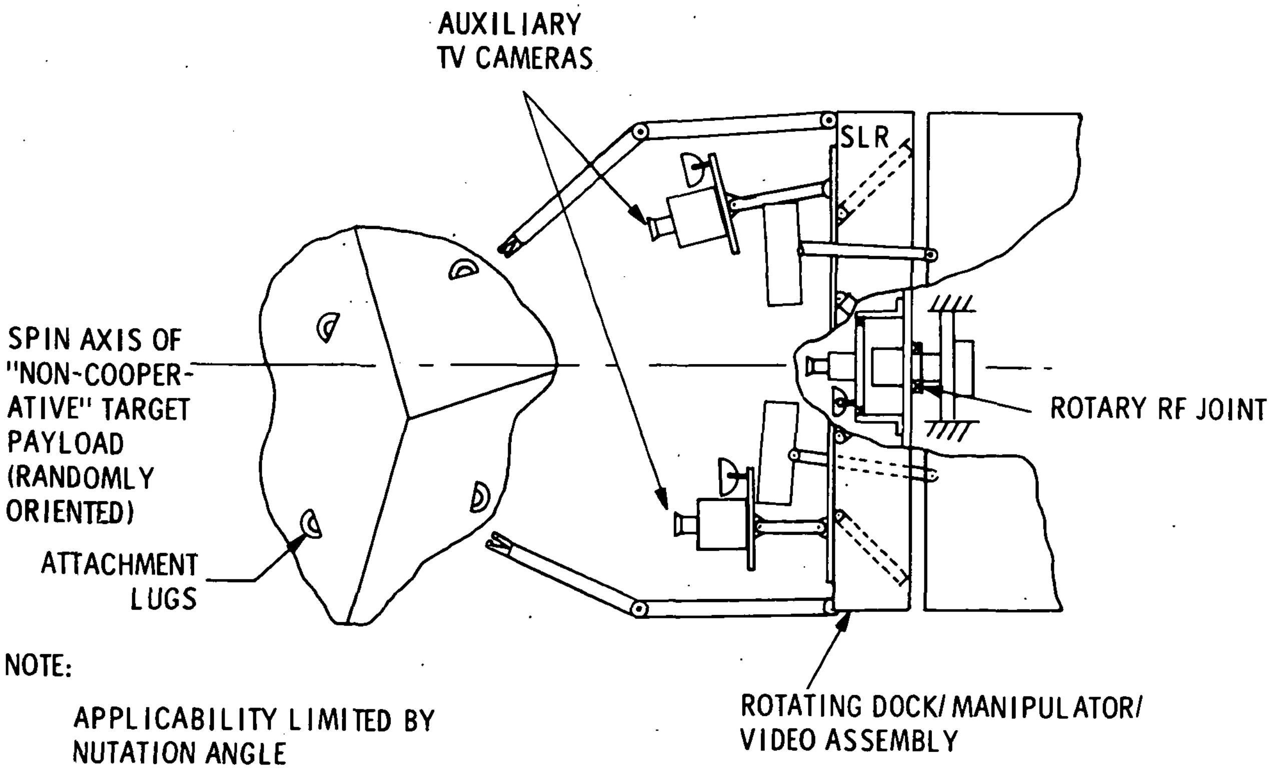


Figure 3-16. Avionics Subsystem Docking Assembly Showing Docking Concept for Randomly Tumbling Loads

REQUIREMENTS

- REACTOR SHUTDOWN
  - LOW TEMPERATURE LIMITS
- ORBITAL OPERATIONS
  - SUN LINE ROTATION
  - MEDIUM CONSTANT POWER DISSIPATION ($\approx$50 W)
  - POWER AVAILABLE
- DOCKING MANEUVERS
  - FIXED ORIENTATION
  - MAXIMUM DISSIPATION REQUIREMENTS ($\approx$350 W)

IMPLEMENTATION

- ISOLATION FROM PC MODULE
- THERMAL LOUVERS FOR VARIABLE EMISSIVITY
- HEAT PIPES FOR CIRCUMFERENTIAL DISTRIBUTION
- HEATER ELEMENTS FOR PROPELLANT TANKS

Figure 3-17. Avionics Subsystem Thermal Control

The wide swing on the thermal dissipation profile complicates the design of this subsystem. The component heat dissipation requirements must be accommodated by the cylindrical shell radiating surface which will have the sun line rotating around it at orbital rate. Thermal isolation from the primary power conditioning assembly is assumed, thereby limiting the problem to rejection of the heat load from the outside shell surface to the external heat sink, taking into account solar flux and earth albedo as required. Power availability for thermal balance is no problem when the reactor is activated, and the most critical problem may be the maintenance of component temperatures above the allowable lower limits during periods when the reactor is not operating. Consequently, these requirements indicate the need for an active or semiactive series element in the thermal control functional system, in order to provide the necessary thermal capacitance that would not be afforded by a purely passive system with its direct path from source to sink. Accordingly, a circumferential heat pipe is assumed to balance the heat load among equipment compartments and to utilize the thermal storage capacity of all components as needed. It may be necessary to incorporate a louver system to permit variation in the α/ϵ ratio (ratio of absorptivity to emissivity), if further analysis so indicates. Heater elements may be required for the hydrazine propellant tanks.

Pending a more detailed study, the assumed implementation of the thermal control subsystem is the incorporation of all three types of components: heat pipes, louvers, and heaters. The future optimization of this subsystem may include integration with the power conditioning module.

3.7 ALTERNATE NEP STAGE CONFIGURATIONS

There are numerous NEP system and mission related elements that significantly effect the final NEP Stage design. The baseline 120 kWe NEP configuration discussed in Section 2.3 reflects the initial mission considerations and preliminary NEP system analysis and integration. In arriving at this preliminary conceptual design, a number of alternate configurations was examined. This section will briefly discuss some of these alternate configurations.

3.7.1 CANDIDATE SYSTEMS

The internal fuel thermionic electric propulsion system for unamnned comet rendezvous and outer planet exploration missions that was defined under Contract JPL-952381 is a side-thrust configuration. The spacecraft is a long, thin cylinder that travels in a direction normal to the spacecraft axis due to the sidewise thrust of a centrally located ion engine bay. The high temperature components (the reactor, neutron shield, and primary heat rejection subsystem) are on the end of the spacecraft, separated from the low temperature components (the electrical power conditioning modules and the payload elements) by the ion engine thruster bay. For purposes of commonality, this same spacecraft design was investigated for geocentric orbit applications.

To investigate the impact of higher operating power levels on mission performance, a 240 kWe end thrust NEP Stage configuration was also evaluated.

3.7.2 PRELIMINARY EXAMPLE DESIGNS

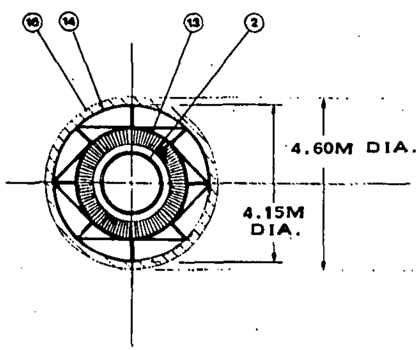
This section presents preliminary example designs of the 120 kWe side thrust NEP Stage configuration and the 240 kWe end thrust stage configuration.

3.7.2.1 <u>120 kWe Side Thurst Configuration</u>

The general arrangement and physical dimensions of the 120 kWe side thrust NEP configuration are shown in Figure 3-18. The overall dimensions of this configuration are 1.6 m in diameter and 19.1 m long. Most of the vehicle length is attributable to the primary heat pipe radiator, the power conditioning radiator, and the ion thruster bay. The configuration shown in Figure 3-18 is designed to be Shuttle launched with a 7.6 m long chemical kick-stage or payload. The side thrust stage must be folded to be transported in the Shuttle cargo bay. To be transported with a 9.1 m long Centaur launch stage would require a slightly shorter configuration at a larger diameter (~1.8 m).

Since the payloads that are being considered for geocentric orbit missions are larger than the net spacecraft scientific payload (for outer planet and comet rendezvous missions) and extend outside the basic diameter of the vehicle, additional reactor radiation shielding is

KEY SYSTEMS PARAMETERS	
POWER TO THRUST S/S	120 kWe @ 23V
SPECIFIC IMPULSE	4000 SEC
SPECIFIC MASS (INCL. NET STAGE)	46 KG/kWe
PROPULSION SYSTEM MASS	5140 KG
NET STAGE MASS	460 KG
ENGINE TYPE	30 CM H_g ION
NUMBER OF THRUSTERS	24



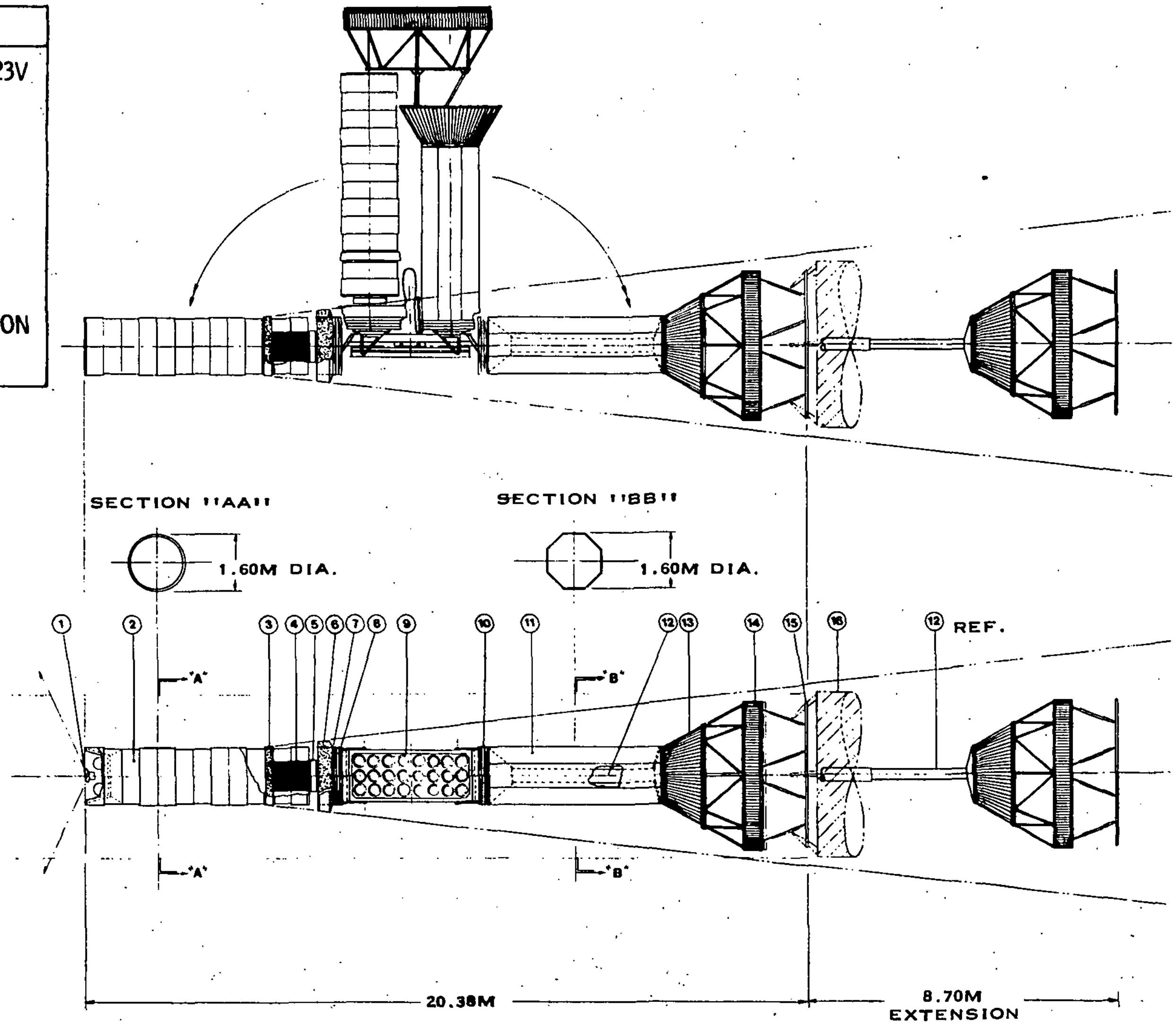


Figure 3-18. General Arrangement of 120 kWe NEP Stage Side Thrust Configuration

necessary. This requirement results from a combination of both direct reactor radiation and scattered radiation off the primary radiator surface. Initial analysis has indicated that radiation scattering off the mercury ion beam is negligible.

In a side thrust vehicle configuration, thrusting must be maintained through the center of gravity. This is particularly affected by the requirement to operate with a wide range of payload mass, and with the payload attached to one end of the vehicle.

Various methods (see Figures 3-19 through 3-21)exist to account for the potential center-of-thrust and center-of-gravity miss-match. One approach is to use a moveable mass system to balance the stage with variable payloads; however, the preferred mode is a spin stabilized center-of-gravity control depicted in Figure 3-21. The NEP Stage is slowly rotated about an axis through the center-of-mass parallel to the thrust axis so that the net torque averages out to zero over one complete spin revolution. This type of maneuver should pose no real problem for the stage, but does somewhat complicate the navigation and control function performed by the avionics subsystem.

A reaction control subsystem must be used in this spin stabilized operational mode to counteract the momentum vector that is generated by spinning the vehicle. The amount of hydrazine propellant required will depend on the spin rate. Assuming a 1 rev/hr vehicle spin rate, the amount of hydrazine for one round trip mission from the reference intermediate parking orbit is estimated to be approximately 20 to 40 kg.

The side thrust NEP Stage configuration of Figure 3-18 is designed with the avionic module mounted on a deployable boom. For geocentric orbit applications, if the stage returns from synchronous orbit with no return payload, the avionics module can be deployed to maintain thrusting through the center-of-gravity with no spin stabilized CG control.

Table 3-12 presents the preliminary mass summary of the 120 kWe side thrust NEP Stage propulsion system. The total mass of the propulsion system is 5140 kg (not including mercury propellant), of which the power subsystem and thrust subsystem contribute 4170 kg and 970 kg, respectively.

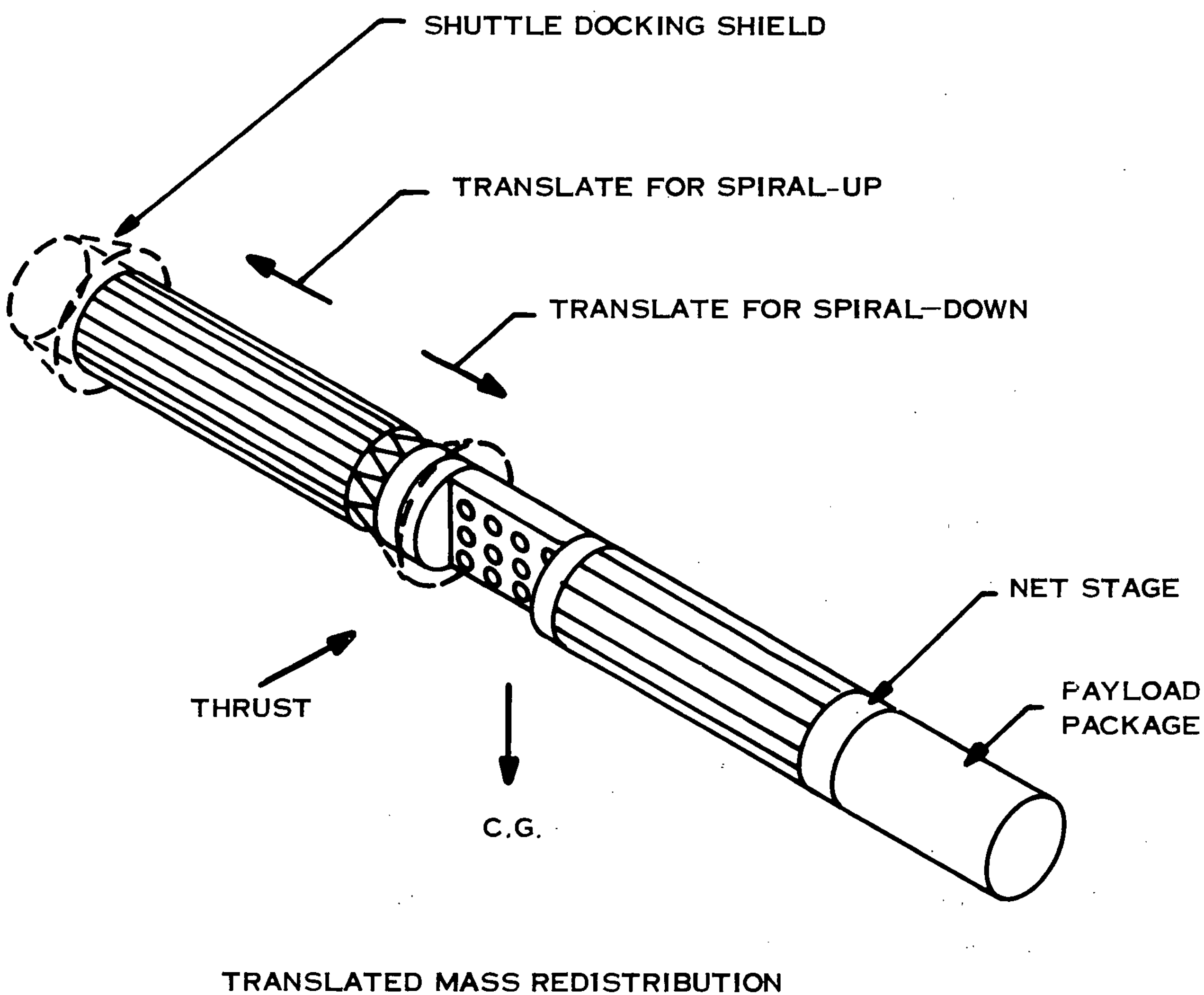


Figure 3-19. Heavy Payload Integration Options

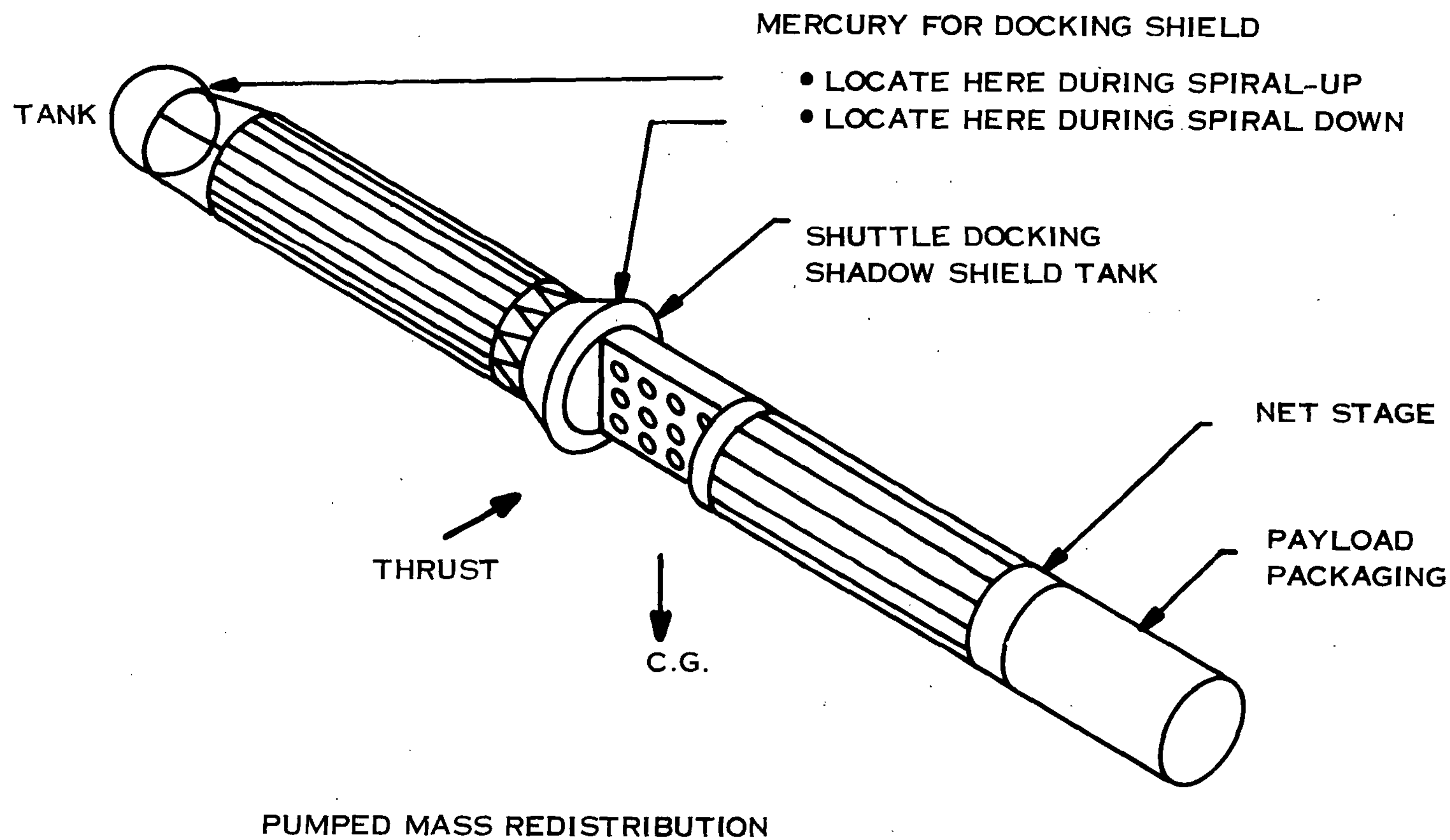
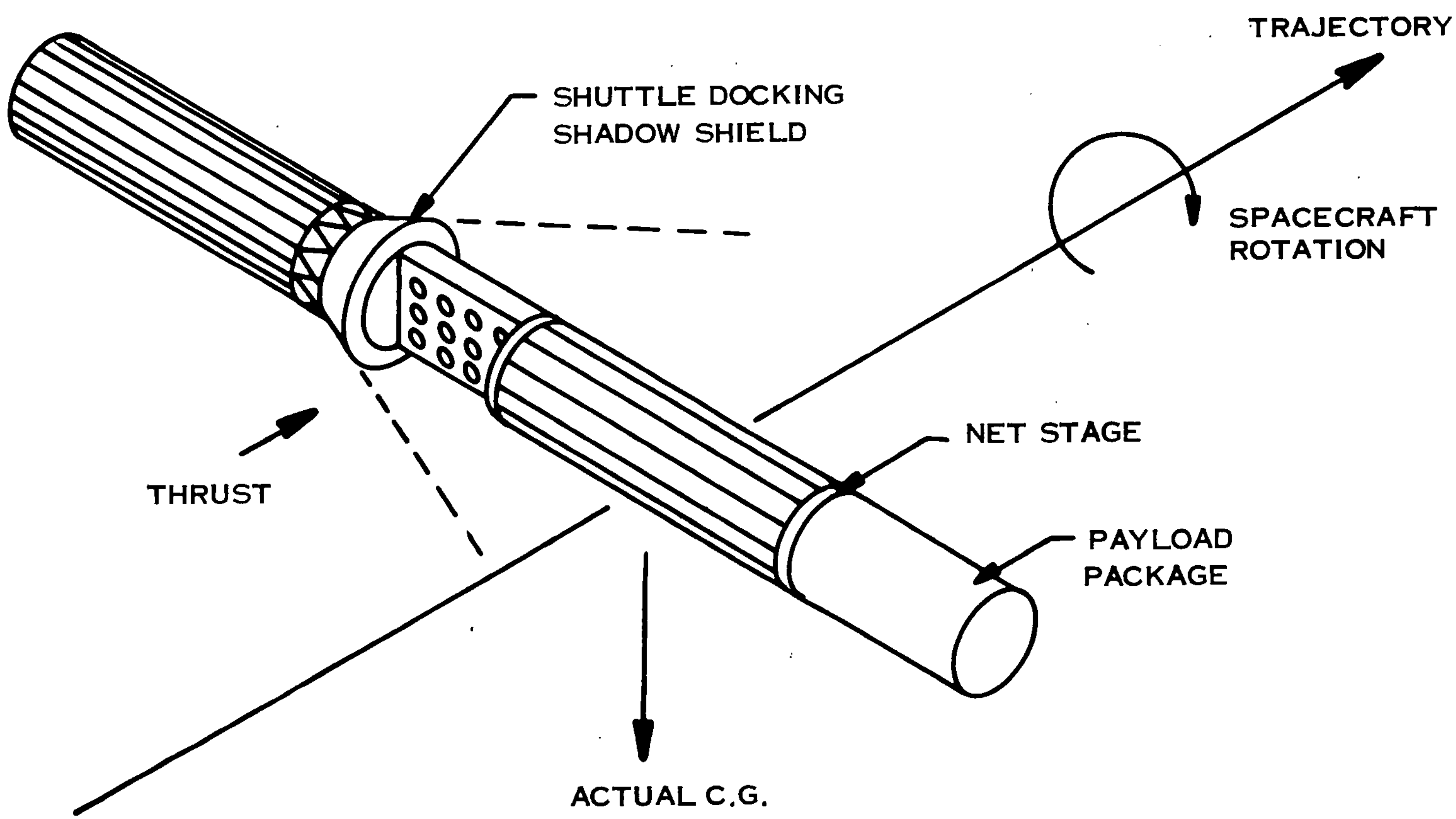


Figure 3-20. Heavy Payload Integration Options



SPIN STABILIZED CENTER-OF-GRAVITY CONTROL

Figure 3-21. Heavy Payload Integration Options

The reactor radiation shielding represents a significant contribution to the specific mass of the side thrust configuration.* The increased shield requirements (relative to the end thrust configuration) are due to the fact that this configuration, designed to shield a 4.6 m diameter payload, is also designed to be packaging in the Shuttle cargo bay with a chemical kick-stage. If this design constraint were removed, the side thrust stage could be increased in length and decreased in diameter resulting in an increase in axial mercury thickness in addition to greater reactor-power conditioning separation distance. This would eliminate the requirement for tungsten permanent gamma shielding and reduce the LiH shield requirements. This would reduce the specific mass from about 46 kg/kWe to about 37 kg/kWe, including the avionics subsystem.

The difference in reactor shielding requirements between this preliminary definition of the 120 kWe side thrust geocentric NEP Stage and earlier the definition of the 120 kWe side thrust interplanetary NEP system are illustrated in Figure 3-22. Increased vehicle diameter was necessitated by the assumed packaging of the NEP Stage in the Shuttle cargo bay. The LiH neutron shielding requirement increased by a factor of 4 kg/kWe. The larger diameter reduced the axial thickness of mercury propellant (relative to the side thrust interplanetary system). Therefore, tungsten permanent gamma shielding is required on this NEP Stage configuration. The permanent gamma shielding contributes 5 kg/kWe to the overall specific mass.

Since the synchronous orbit payloads may be as large as 4.6 m in diameter, shadow shielding and shielding for neutron back-scatter is required on the multi-mission NEP Stage. This shielding required for the large diameter payloads contributes an additional 5 kg/kWe.

If the side thrust NEP Stage packaging provides for three side-by-side folded segments, the vehicle diameter could be reduced. This would result in greater mercury shielding thickness,

*The radiation shield on the side thrust configuration shown in Figure 3-18 (and the 240 kWe end thrust configuration shown in Figure 3-23) is designed to shield the power conditioning electronics to 10^{12} nvt ($E_n \geq 1$ MeV) and 10^7 rads γ. These configurations were designed prior to the establishment of the photon dose limit at 10^6 rads for which the reference 120 kWe end thrust NEP Stage is designed.

1971 NEP SIDE THRUST INTERPLANETARY S/C

SHIELD	DIAMETER (M)	THICKNESS (M)	MASS (KG)	$\Delta\alpha$ @ 120 kWe (KG/kWe)
NEUTRON (LiH)	1.14	0.51	380	3
GAMMA (Hg)	1.14	0.28	3660	-
GAMMA (W)	NONE			

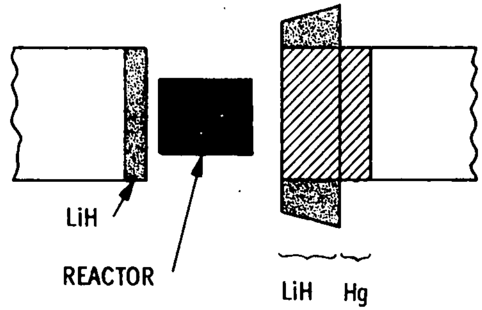
SIDE THRUST NEP TUG

CONSTANT DIAMETER SHIELDING FOR PC ELECTRONICS

SHIELD	DIAMETER (M)	THICKNESS (M)	MASS (KG)	$\Delta\alpha$ @ 120 kWe (KG/kWe)
NEUTRON (LiH)	1.6	0.50	725	7
GAMMA (Hg)	1.6	0.053	1370	-
GAMMA (W)	1.6	0.016	600	5

SHIELDING REQUIRED FOR 4.6M (15 FT) DIAMETER P/L

SHIELD	THICKNESS (M)	MASS (KG)	$\Delta\alpha$@ 120 kWe (KG/kWe)
NEUTRON (LiH) SCATTER (D = 1.6 M)	0.2	290	2
NEUTRON (LiH) SCATTER (CIRCUMFERENTIAL)	0.5	305	3



- INCREASED SHIELDING
  - LARGER DIAMETER DUE TO SHUTTLE INTEGRATION WITH CENTAUR/CHEMICAL TUG
  - LARGE PAYLOAD DIAMETER
- NEP TUG DIAMETER CAN BE REDUCED IF NOT PACKAGED WITH CENTAUR/CHEMICAL TUG ($\alpha \sim$ 36-37 KG/kWe)

Figure 3-22. Reactor Shielding for NEP Side Thrust Configurations

greater separation distance between the reactor and power conditioning electronics, and reduced LiH shield mass. If the NEP Stage diameter were reduced to about 1.2 m, the stage specific mass could be reduced to 36-37 kg/kWe, including the avionics module.

3.7.2.2 240 kWe End Thrust Configuration

The general arrangement and physical dimensions of the 240 kWe end thrust NEP Stage configuration are shown in Figure 3-23. The 240 kWe NEP Stage configuration employs two flashlight thermionic reactors, each operating at approximately 1540 kWt to deliver a total of 240 kWe at 23 volts to the thrust subsystem.

The 240 kWe NEP Stage is basically the same general configuration as that of the reference 120 kWe Stage, only larger.* The radiator areas are about twice as large as for the 120 kWe design, and the thruster array contains forty-eight 30 cm ion engines as compared to twenty-four for the reference stage.

The 240 kWe NEP Stage is designed to perform the geocentric orbit mission with no chemical assist. To do this, the Stage must ascend and descend through the Van Allen radiation belt on each synchronous orbit payload mission. To protect the power conditioners from the electron and proton radiation, the electronics may be further shielded by beryllium covers.

The larger system would affect the design of the avionics subsystem considerably if it were considered necessary to fold the vehicle in the Shuttle cargo bay as opposed to usage of a telescoping deployment system. This would result in either:

1. The avionics subsystem (and payload) not being concentric with the remainder of the NEP Stage.
2. The avionics subsystem being concentric with the rest of the NEP Stage but would be of much smaller diameter than the payload.

*A conical primary radiator is employed to achieve the minimum length vehicle for Shuttle packaging considerations.

KEY SYSTEM PARAMETERS	
POWER TO THRUST S/S	240 kWe @ 23V
SPECIFIC IMPULSE	4000 SEC
SPECIFIC MASS (INCL. NET STAGE)	37 KG/kWe
PROPULSION SYSTEM MASS	8400 KG
NET STAGE MASS	460 KG
ENGINE TYPE	30CM Hg ION
NUMBER OF THRUSTERS	48

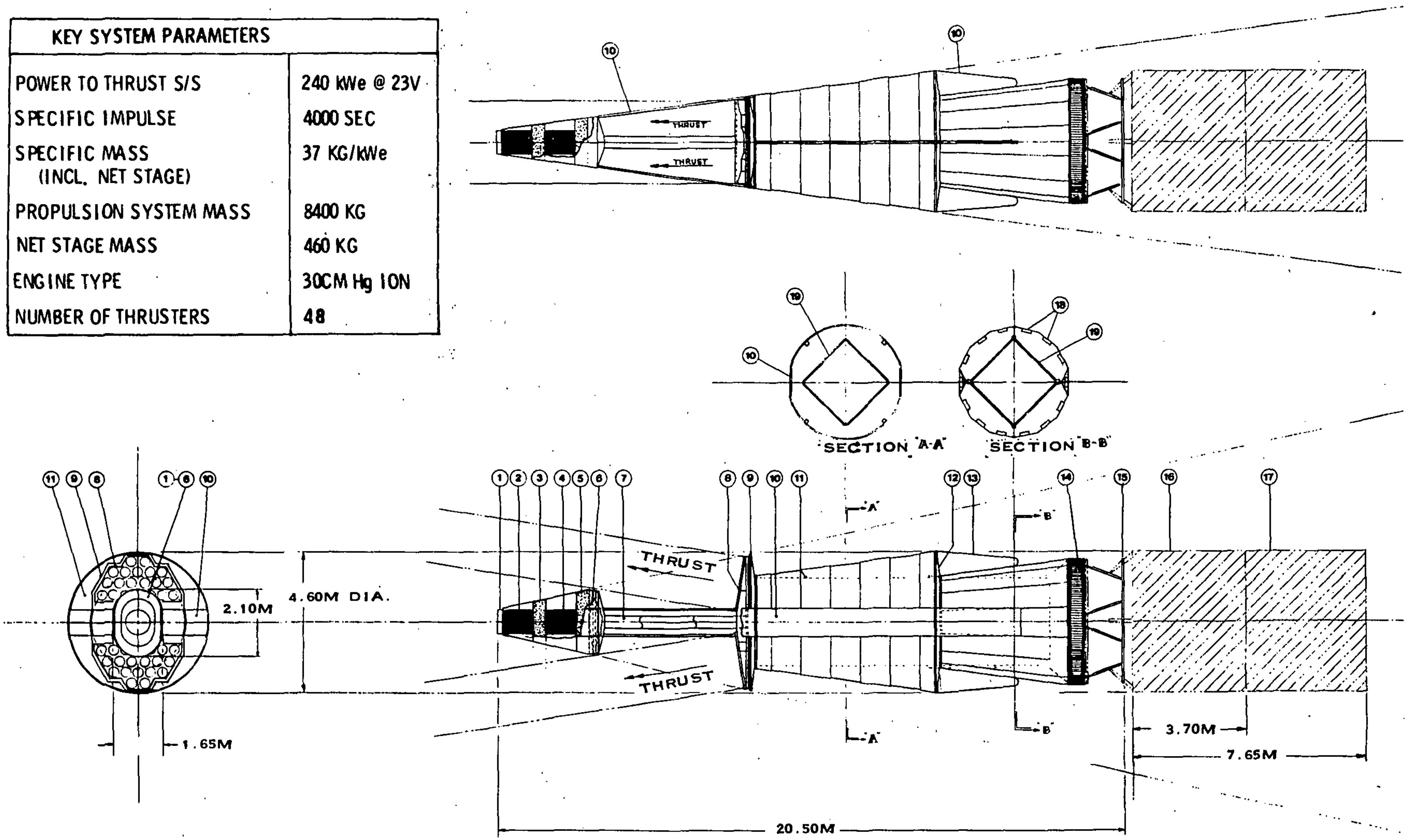


Figure 3-23. General Arrangement of 240 kWe End Thrust NEP Stage (Alternative Configuration)

In the former case, the attitude control thrusters and ion engines would be offset and non-symmetrical with respect to the CM, and the control problem would be somewhat complicated. In the latter case, the payload could well block the view of the avionics subsystem sensors, thrusters, thermal radiating surfaces, etc., thus creating design problems in this area.

Folding of the vehicle also results in a longer, smaller diameter NEP Stage with more inherent structural flexibility with a smaller ratio of volume to structural surface area. These factors may not be a disadvantage, however, since the load environment (other than boost) is essentially negligible and the volume requirements are very small. It could be that the longer, smaller diameter system could have less structural weight than a shorter, stubbier design, if a minimum structural gauge situation develops.

It would appear that a larger diameter NEP Stage with a telescoping deployment system would be more advantageous than a foldable system. It would appear to make better use of the Shuttle cargo bay volume, and would essentially be a scaled-up version of the 120 kWe Stage from the standpoint of avionics subsystem design. However, the detailed trade studies required for the full evaluation would have to be identified and performed.

Table 3-12. Mass Summary of 120 kWe NEP Stage Propulsion System
Side Thrust Configuration

Power Subsystem

Component	Mass (kg)
Reactor	1440
Heat Rejection Subsystem	570
Neutron Shield	1320
Tungsten Gamma Shield	600
Hotel PC	50
Hotel PC Radiator	20
Startup Auxiliary Power Supply	50
Structure	120
Total	4170

Thrust Subsystem

Component	Mass (kg)	
Thruster Array	305	
PC Modules	300	
PC Radiator	125	
Power Transmission Cable	170	
Mercury Propellant Tanks and Distribution	40	Preliminary Mass Summary Mass Reduction Techniques Have Been Identified
Structure	30	
Total	970	
Mercury Propellant	1370	
Total Propulsion System Mass (Does not Include Hg Propollant)	5140 kg	

SECTION 4

MISSION ANALYSIS

This section describes the mission analysis effort and presents the mission profiles for nuclear electric propulsion interplanetary and geocentric earth orbit applications.

4.1 INTERPLANETARY MISSIONS

The interplanetary mission analysis effort is directed toward the definition of a baseline Comet Halley rendezvous mission and the selection and definition of at least one baseline outer planet mission. These mission characteristics are employed in the preliminary definition of the multi-mission NEP Stage requirements, including its power level.

Key assumptions particular to the interplanetary mission analysis effort are delineated in Table 4-1. High thrust earth escape is employed using the Centaur D-1T. Launch to earth orbit is accomplished by either the Space Shuttle or a Titan class launch vehicle. Based on preliminary evaluations of typical Net Spacecraft component mass requirements, * the mass of the Net Spacecraft is held constant at 700 kg.

The NEP Stage propulsion system specific mass employed in the interplanetary mission analysis is a particular function of the electric power delivered to the thrust subsystem, P_e. This relation, given in Table 4-1, is based on previous NEP Stage mass studies.

The Comet Halley rendezvous mission is specified as a requirement for the multi-mission NEP Stage. The bulk of the mission analysis is therefore devoted to defining at least one outer planet mission to assist in the definition of the multi-mission stage. The candidate outer planet missions evaluated in this study are listed in Table 4-2. These missions include Jupiter, Saturn, and Uranus orbiters, and a Neptune flyby.

*At the time the interplanetary mission analysis was performed, the NEP Stage was defined to consist of a propulsion system (thrust subsystem and power subsystem) and a propellent system. The Net Spacecraft included the science, commmunications, data handling, and stage control systems. The currently defined NEP system (see Section 3) consists of a thrust subsystem, power subsystem, propellant subsystem, and avionics subsystem. The interplanetary payload contains just the science required for the performance of the mission. The communications, data handling, and stage control functions are all contained in the avionics subsystem.

Table 4-1. Mission Analysis Particular Assumptions

High Thrust Earth Escape
Shuttle/Centaur (Baseline)
Titan/Centaur
Space Shuttle
27,000 kg Payload
270 nm Circular Orbit
Centaur
470 Sec Specific Impulse
10.2 Percent Tankage Factor
700 kg Net Spacecraft
Baseline Propulsion System Specific Mass
α [kg/kWe] = 258 P_e Exp (-0.474)

Figure 4-1 shows the performance characteristics for the NEP Stage propulsion system. The variation of propulsion system specific mass with the electric power delivered to the thrust subsystem, P_e, is based on earlier thermionic NEP spacecraft design studies (References 4-1, 4-2) completed at power levels of approximately 75 kWe, 120 kWe and 275 kWe. Curve A is employed in the baseline studies presented in this report. Effects of changing the slope of the baseline curve, either a high power bias, or a low power bias are also assessed. For example, the high power bias, below 100 kWe, acts to increase trip time and propulsion time for a fixed payload. Curve D presents the specific mass relation specified by JPL for trajectory analyses.

4.1.1 SHUTTLE-CENTAUR LAUNCH TO EARTH ESCAPE

Results of the mission analysis* using the Shuttle/Centaur launch vehicle are presented in Table 4-3 for optimum power Jupiter and Saturn orbiter missions. Optimum power is that power level which results in minimum mission energy for a given trip time. An off-optimum power level can result in greater Net Spacecraft mass, but total mission energy in terms of propulsion time, trip time, and specific impulse will be higher.

*Performed with trajectory code supplied by JPL (Reference 4-3).

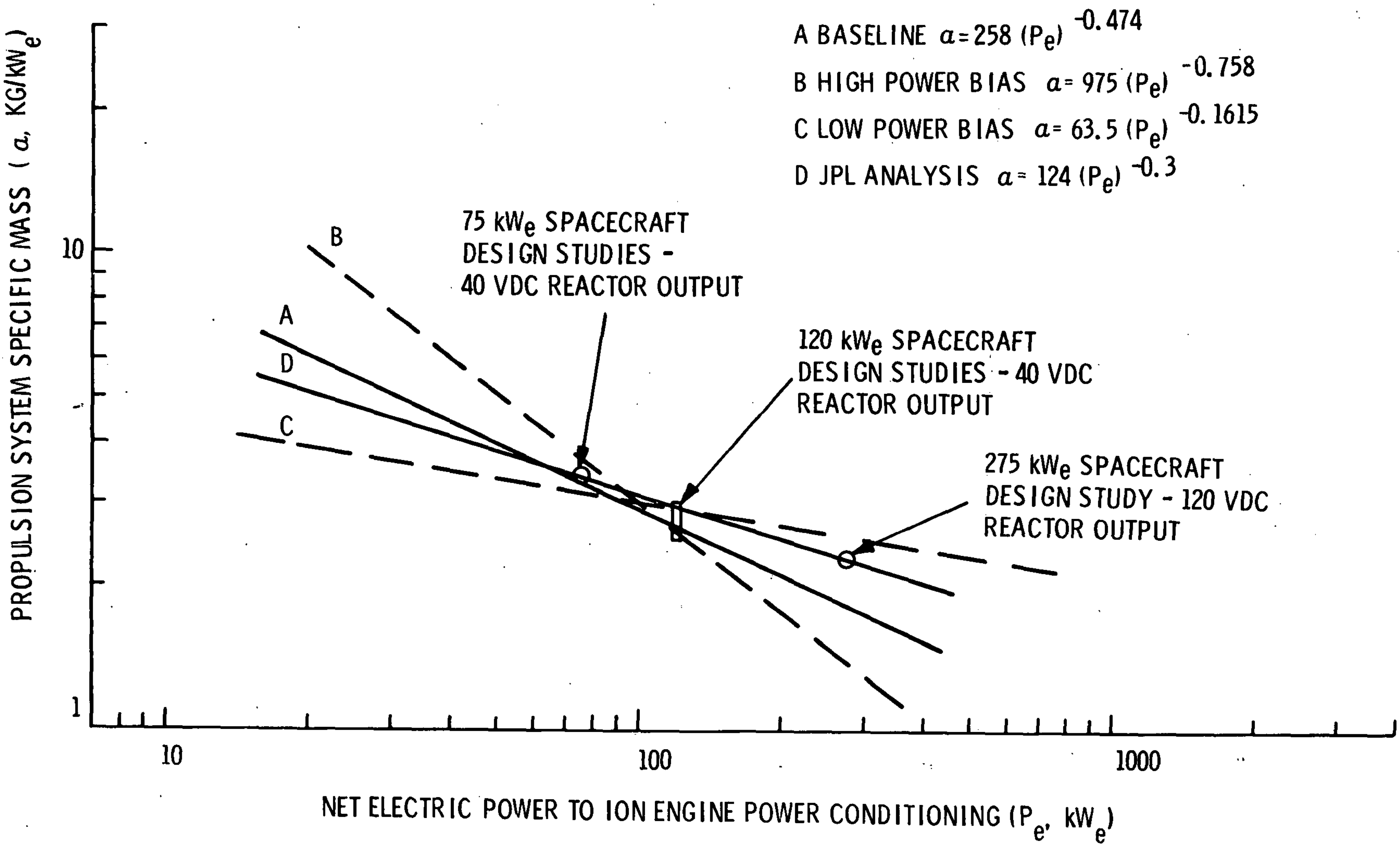


Figure 4-1. NEP Stage Propulsion System Performance Characteristics

Table 4-2. Candidate Outer Planet Missions High Thrust Earth Escape/Low Thrust Capture

Planet	Satellite in Capture Orbit	Capture Orbit Planetary Radii	Launch Vehicles
Jupiter	I_o	5.9	Space Shuttle/Centaur and TIIID7/Centaur
	Ganymede	15.0	Space Shuttle/Centaur and TIIID7/Centaur
	Callisto	26.3	Space Shuttle/Centaur and TIIID7/Centaur
Saturn	Tethys	4.9	Space Shuttle/Centaur and TIIID7/Centaur
	Titan	20.4	Space Shuttle/Centaur and TIIID7/Centaur
Uranus	Titania	18.5	Space Shuttle/Centaur and TIIIL4/Centaur
Neptune	--	Flyby	Space Shuttle/Centaur and TIIIL4

Table 4-3. Baseline Mission Performance with Space Shuttle Launch of Optimum Power NEP Stage

Mission	Jupiter R=5.9	Jupiter R=26.3	Saturn R=4.9	700 kg Net Spacecraft Saturn R=20.4
Launch Vehicle	Shuttle			
P_e (kW_e)	145	158	125	130
α (kg/kW_e)	24.4	23.6	26.3	25.8
Trip Time (Days)	780	575	1150	985
Specific Impulse (sec)	3700	3000	4000	3500
Departure Hyperbolic Velocity (km/sec)	3.0	3.8	3.6	4.1
Capture Time (Days)	96	8.0	68	9.0
Propulsion Time (hours)	12,400	8,400	17,000	14,000

Variations in power level between the example missions, as well as specific impulse is inconsistent with the use of a single multi-mission spacecraft design to perform all missions. Different power levels require different size propulsion systems, and different specific impulses require changes in the main power conditioning and/for ion engines, even for a fixed size spacecraft.

Table 4-4 shows the effect on mission performance of making the NEP Stage power level constant at 120 kWe. It is seen that fixing the power level at 120 kWe for the NEP Stage (launch by the Shuttle/Centaur) results in essentially no change in performance relative to an optimum power spacecraft, since only a small change in power is involved (less than 20 percent). However, the variation in specific impulse between the candidate missions prohibits delineation of a single stage to perform all missions. Different specific impulses require different high voltage supplies to the ion engines, and can require the separate development and qualification of the main power conditioning for each mission.

Table 4-4. Baseline Mission Performance with Space Shuttle Launch of 120 kW_e NEP Stage

Mission	Jupiter R=5.9	Jupiter R=15	Jupiter R=26.3	Saturn R=4.9	Saturn R=20.4	Uranus R=18.5	Neptune Flyby
Trip Time (Days)	800	635	585	1150	1000	1720	1370
Specific Impulse (sec)	4400	4200	4100	5000	5000	5800	6200
Departure Hyperbolic Velocity (km/sec)	6.0	6.4	6.5	5.9	6.2	5.6	6.4
Capture Time (days)	121	35	12	78	11	11	–
Propulsion Time (hours)	12,600	9500	8500	16,800	14,000	23,500	16,000

Launch Vehicle: Shuttle/Centaur D-1T
Net Spacecraft: 700 kG
P_e = 120 kW_e
α = 26.8 kG/kW_e

As indicated in Table 4-5, a single NEP Stage design can be defined which will perform multiple outer planet missions, as well as the Comet Halley rendezvous. Fixed power level, specific impulse, specific power, and Net spacecraft mass identify the electric propulsion propellant inventory as the only variable that can affect the NEP Stage design. The range of propellant inventories shown for these candidate missions (3300 kg to 4500 kg) can be readily accommodated within a single stage design by sizing the tank system to accommodate the largest mercury inventory required for a family of missions. The tank structure weight penalty necessary to incorporate this feature will be negligible.

The establishment of a fixed 120 kWe power level, rather than the optimum for each mission not only assists in providing the same propulsion system with a multi-mission capability, but improves mission performance. The additional mission energy required for employing an off-optimum propulsion system is obtained from higher specific impulse, which is established at 5000 seconds for the baseline interplanetary missions.

Figure 4-2 presents the electric propulsion system NET spacecraft (payload) performance for a Shuttle/Centaur D-1T launched 120 kWe NEP Stage with 5000 seconds specific impulse for Comet Halley rendezvous and outer planet missions. The figure shows that a small increase in trip time will result in large increases in Net spacecraft mass because of the steep slope of the curves. For the baseline Comet Halley rendezvous mission, if the trip time is allowed to increase from 900 days to 1100 days (22 percent increase), the Net spacecraft mass can be increased from 700 to 1260 kg (80 percent increase).

The effect of an arbitrary increase of 5 kg/kWe (approximately 20 percent) in propulsion system specific mass, from the baseline value of 26.8 kg/kWe corresponding to a P_e of 120 kWe, is shown in Table 4-6. This increase reduces the allowable Net spacecraft mass by approximately 600 kg if mission time is held constant. In order to maintain the baseline value of 700 kg, the trip time must be increased by approximately 100 days for the close Jupiter and Saturn orbiters, 150 days for the Neptune flyby, and 200 days for the Uranus orbiter. Propulsion time increases to about 2000 hours are also required.

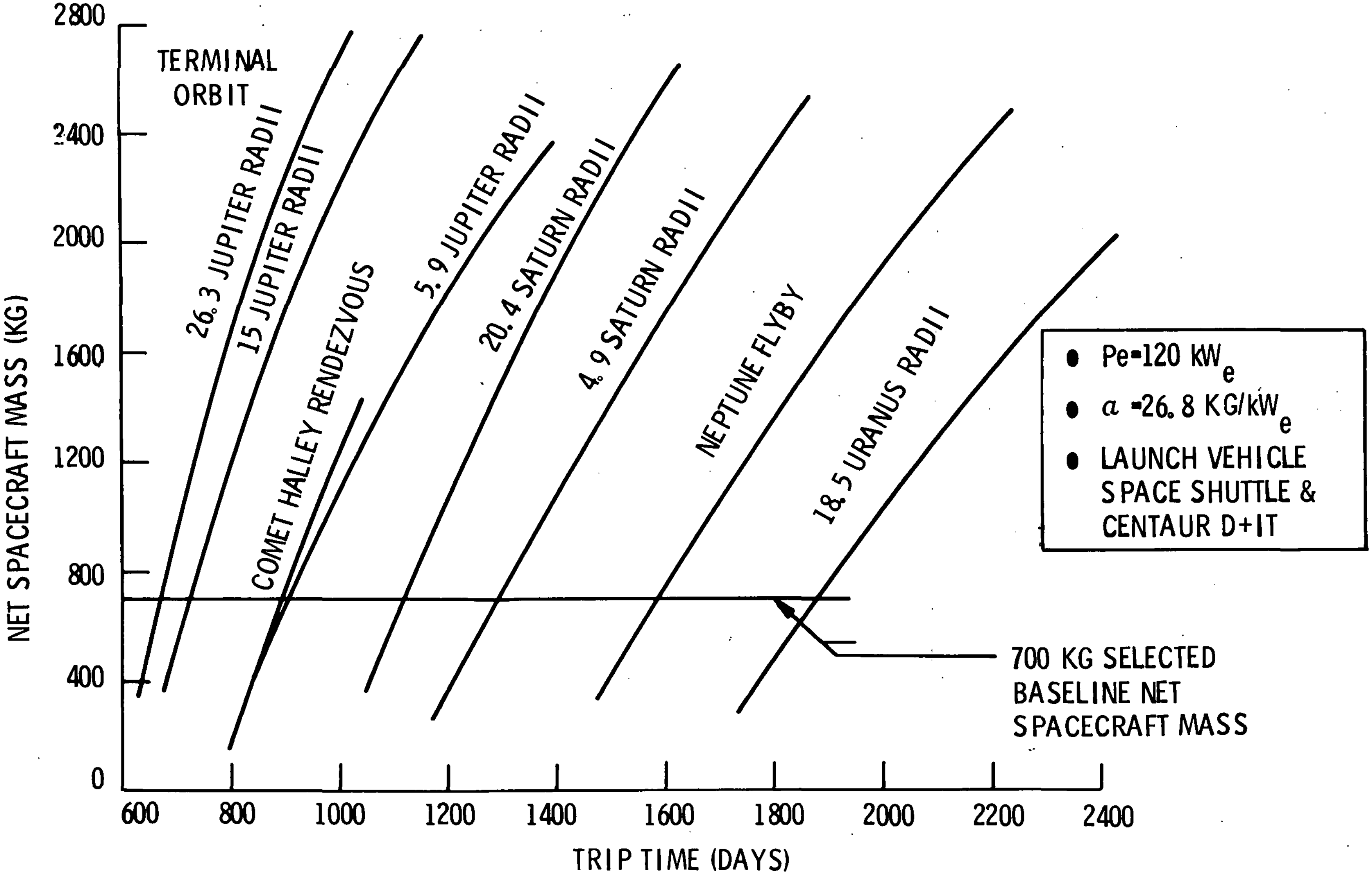


Figure 4-2. Effect of Trip Time on Net Spacecraft Mass

Table 4-5. Mission Performance of 120 kW_e NEP Stage for Outer Planet and Comet Halley Missions

Mission	Comet Halley Rendezvous	Jupiter R=5.9	Jupiter R=15	Jupiter R=26.3	Saturn R=4.9	Saturn R=20.4	Uranus R=18.5	Neptune Flyby
Trip Time (days)	900	900	700	650	1320	1120	1950	1650
Departure Hyperbolic Velocity (km/sec)	2.5	2.9	3.5	3.8	2.9	3.3	2.8	3.6
Mercury Propellant Weight (kg)	4500	4200	3700	3300	4200	3800	4300	3600
Capture Time (days)	--	158	45	18	95	12	16	--
Propulsion Time (Hours)	18,000	14,000	12,000	11,500	17,000	15,200	21,000	15,000

Constant Mission Parameters

• Launch Vehicle	Space Shuttle/Centaur D-1T
• Power to Thrusts Subsystem, P_e	120 kW_e
• Propulsion System Specific Mass	27 kg/kW_e
• Specific Impulse	5000 Seconds
• Net Spacecraft	700 kg

Table 4-6. Effect of 5 kg/kW Specific Mass Increase on Mission Performance

Mission	Jupiter R=5.9	Jupiter R=15	Jupiter R=26.3	Saturn R=4.9	Saturn R=20.4	Uranus R=18.5	Neptune Flyby
Trip Time (days)	900	700	650	1280	1100	1900	1500
Specific Impulse (sec)	4400	4200 5 8	4200 5.8	5100	5000	5800	5900
Departure Hyperbolic Velocity (k/m/sec)	5.3	5.8	5.8	5.0	5.4	4.5	5.6
Capture Time (days)	145	45	16	78	12	16	--
Propulsion Time (hours)	13,800	10,300	9400	16,800	15,300	25,700	18,000

Launch Vehicle: Shuttle/Centaur D-1T
Net Spacecraft: 700 kg

$$P_e = 120 \text{ kW}_e$$
$$= 31.8 \text{ kg/kW}_e$$

Early IOC (early 1980's) NEP systems are characterized by life-limited propulsion systems. Table 4-7 shows the effect of constraining propulsion times to about 10,000 hours or less for the 120 kWe NEP Stage for a Shuttle/Centaur D-1T launch. Specific impulse decreases to about 4200 seconds, and trip time increases directly with the decrease in propulsion time. For the Jupiter orbiter mission of 5.9 Jupiter radii, the reduction of propulsion time from 12,600 hours (see Table 4-4) to 10,500 hours increases trip time only 25 days. The trip time increase for a reduction is propulsion time from 16,800 hours to 10,000 hours for a Saturn orbiter mission of 4.9 radii is 170 days. A portion of the increased trip time can be recovered

by increasing the specific impulse back to 5000 seconds. However, it is important to note that all the candidate outer planet missions evaluated can be performed with 10,000 full power hours of propulsion time or less.

4.1.2 TITAN CENTAUR LAUNCH MISSION PERFORMANCE

Characteristics of mission performance for the baseline outer planet missions using an optimum power NEP Stage launched by Titan/Centaur vehicles are shown in Table 4-8. Optimum power, which is primarily a direct function of launch vehicle capability, ranges from 63 to 75 kWe for the Jupiter and Saturn orbiter missions, is 160 kWe for the Uranus orbiter, and is 95 kWe for the Neptune flyby. The effect on mission performance, if power level is allowed to vary such that minimum mission energy is achieved, is that trip time increases and propulsion times increase, except for the Uranus orbiter when the optimum power level is greater than the 120 kWe constrained value. The most significant reduction in mission energy with an optimum power propulsion system is in specific impulse, which decreases almost 2000 seconds for the Jupiter and Saturn orbiter missions. Specific impulse decreases 700 seconds for the Neptune flyby mission and increases by 300 seconds for the Uranus orbiter because of the increase in power level from 120 kWe to the optimum value of 160 kWe. Consequently, the establishment of a fixed, 120 kWe power level, rather than the optimum for each mission performance in terms of trip time and propulsion time. The additional mission energy required for employing an off-optimum propulsion system is obtained from higher specific impulse, which is established at 5000 sec for all the baseline interplanetary missions.

Table 4-9 presents the mission performance of a 120 kWe NEP Stage launched by a Titan/Centaur for the candidate outer planet missions. For a mission to the Jovian orbit of 5.9 radii, total trip time of 960 days, propulsion time of 14,600 hours, and specific impulse of 5500 sec are required. Descent to the 5.9 radii orbit is initiated 774 days into the mission. Similarly, for a Saturn orbiter mission of 4.9 radii, a trip time of 1360 days, a propulsion time of 19,400 hours, and a specific impulse of 6400 seconds are required. The Uranus orbiter can be accomplished with a moderate specific impulse of 4700 seconds and trip time of 1670 days, but propulsion time is 24,400 hours, which is in excess of reactor lifetime

Table 4-7. 10, 000 Hour Propulsion Time Constraint Shuttle/Centaur Launched 120 kW_e NEP Stage

Mission	Jupiter R=5.9	Jupiter R=15	Jupiter R=26.3	Saturn R=4.9	Saturn R=20.4	Uranus R=18.5	Neptune Flyby
Trip Time (days)	825	635	585	1320	1040	2250	1500
Specific Impulse (sec)	4200	4200	4100	4200	4300	4000	4200
Departure Hyperbolic Velocity (km/sec)	6.2	6.4	6.5	6.2	6.5	6.1	6.7
Capture Time (days)	125	35	12	100	11	25	--
Propulsion Time (hours)	10, 500	9500	8500	10, 000	10, 000	10, 000	10, 000

Launch Vehicle: Shuttle/Centaur D-1T
Net Spacecraft: 700 kg
$P_e = 120\ kW_e$
$\alpha = 26.8\ kg/kW_e$

projected for an early generation interplanetary mission. A Neptune flyby mission can be achieved with a trip time of 1550 days, a propulsion time of 18, 800 hours, and a specific impulse of 5700 seconds.

In order to decrease the specific impulse and propulsion time requirements for the missions indicated, a launch vehicle of greater payload capability may be employed, or propulsion time may be constrained at the expense of increased trip time and departure hyperbolic excess velocity. The latter factor is not a penalty, but merely indicates the increased use of the high thrust system from the optimum amount to accomplish the mission. It is noted that near-constant specific impulse for all missions facilitates the design of a multi-mission stage.

For each of the baseline outer planet missions evaluated, available net payload as a function of trip time has been evaluated for a 120 kWe NEP Stage that is launched by a Titan/Centaur class launch vehicle. This effect of trip time on Net spacecraft mass is shown in Figure 4-3.

Table 4-8. Baseline Mission Performance of Optimum Power NEP Stage Launched by Titan Centaur Family

Mission	Jupiter R=5.9	Jupiter R=15	Jupiter R=26.3	Saturn R=4.9	Saturn R=20.4	Uranus R=18.5	Neptune Flyby
Launch Vehicle	Titan III D7/Centaur	Titan III D7/Centaur	Titan III D7/Centaur	Titan III D7/Centaur	Titan III D7/Centaur	Titan III L4/Centaur	Titan III
P_e (kW_e)	71	75	74	63	65	160	95
α (kg/kW_e)	34.2	33.2	33.3	36.2	35.6	23	29.8
Trip Time (days)	1090	830	750	1390	1260	1700	1675
Specific Impulse (sec)	3600	3400	3240	4000	4000	5000	5000
Departure Hyperbolic Velocity (kW sec)	1.8	2.2	2.6	2.2	2.0	2.0	2.1
Capture Time (days)	185	55	19	35	< 1	10	--
Propulsion Time (hours)	16,6000	12,500	11,000	20,000	19,200	24,200	25,000

Table 4-9. Baseline Performance of 120 kW_e NEP Stage Launched by Titan/Centaur Family

Mission	Jupiter R=5.9	Jupiter R=15	Jupiter R=26.3	Saturn R=4.9	Saturn R=20.4	Uranus R=18.5	Neptune Flyby
Launch Vehicle	Titan III D7/Centaur	Titan III D7/Centaur	Titan III D7/Centaur	Titan III D7/Centaur	Titan III D7/Centaur	Titan III L4/Centaur	Titan III L4
Trip Time (days)	960	750	700	1360	1170	1670	1550
Specific Impulse (sec)	5500	5000	5000	6400	6000	4700	5700
Departure Hyperbolic Velocity (kW sec)	2.3	2.9	3.1	2.4	2.7	2.2	2.5
Capture Time (days)	146	44	16	95	12	9	--
Propulsion Time (hours)	14,600	11,000	10,000	19,400	16,000	24,400	18,800

P_e = 120 kW_e

α = 26.8 kg/kW_e

700 kg Net Spacecraft

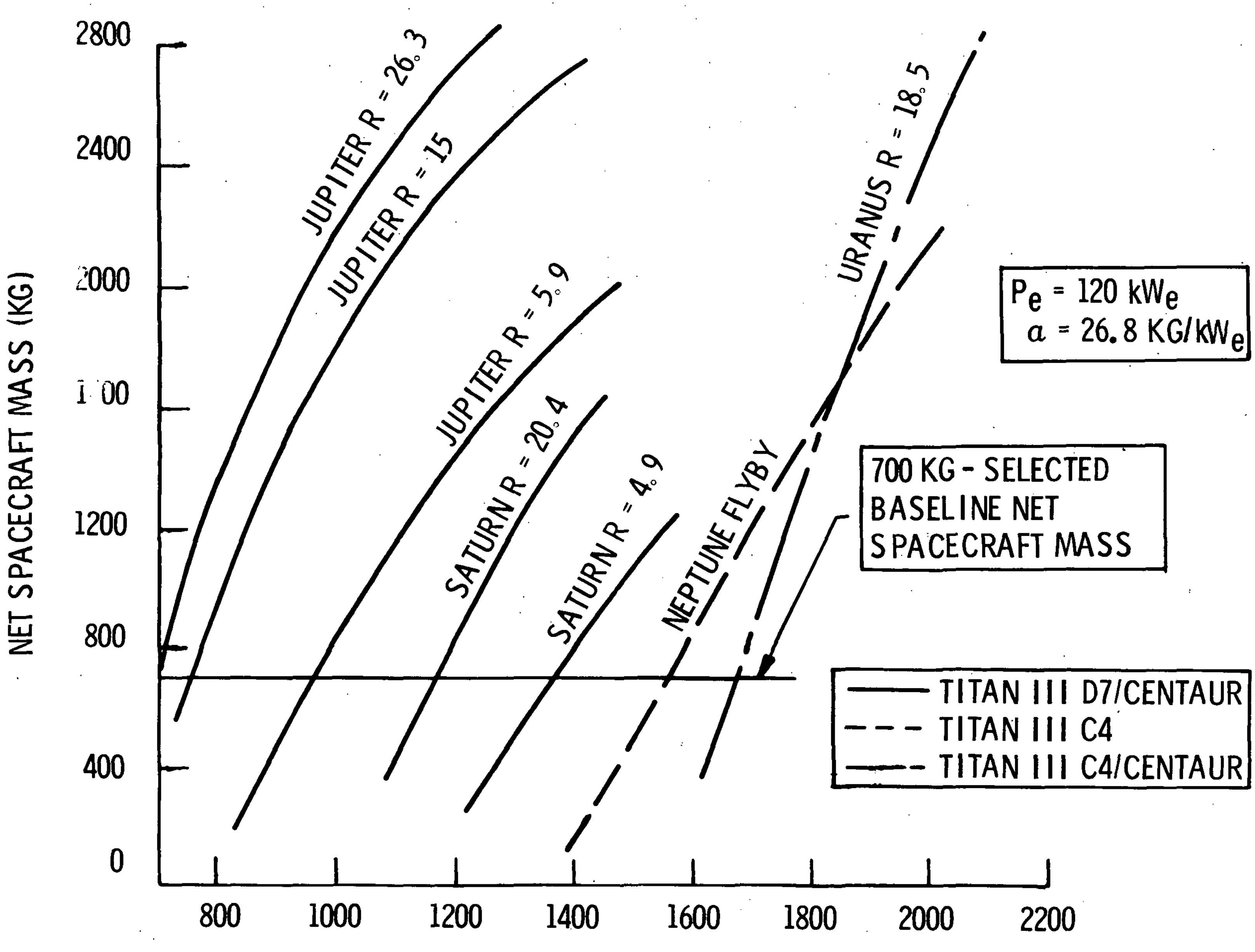


Figure 4-3. Effect of Trip Time on Net Spacecraft Mass

Comparing the data in Figure 4-3 with that in Table 4-8, it is seen that greater payload capability is available for a 120 kWe NEP Stage than with the optimum power NEP Stage. However, the 120 kWe NEP Stage requires either significantly larger specific impulse in the low thrust propulsion system or longer propulsion times.

4.1.3 BASELINE MISSION SELECTION

The results of the interplanetary mission analysis lead to the selection of a power level in the 100 kWe to 120 kWe range for the multi-mission NEP Stage. This value results in decreased trip times relative to lower, optimum power systems and is compatible with both U-235 fueled external fuel and internal fuel in-core thermionic reactor design concepts. A value of 120 kWe is also compatible with Shuttle integration.

In addition to the baseline Comet Halley rendezvous mission, the tight Jupiter orbiter is selected as the baseline outer planet mission. It is more difficult than high radii orbiters and of potentially greater scientific value. These two missions will be employed in the delineation of mission operation events.

Figure 4-4 depicts the baseline Comet Halley Rendezvous mission. This mission, with a trip time of 900 days, requires a low thrust propulsion time of 18,000 hours and an initial hyperbolic excess velocity of 2.5 km/sec. For this mission, the spacecraft is launched to earth escape in May 1983, and comet rendezvous is in December 1985, which is fifty days before perihelion. This marks the beginning of approximately 100 days of scientific observation within the environs of the comet.

The Comet Halley mission is characterized by an accelerate-decelerate-accelerate electric propulsion thrust profile. The comet orbit is retrograde and slightly out of the ecliptic. This feature is exaggerated in the figure.

Net spacecraft mass as a function of trip time for the Comet Halley rendezvous mission is presented in Figure 4-5 for both spiral-out and direct injection earth escape modes. These data were generated by NASA-Ames. The baseline Comet Halley mission is indicated. The

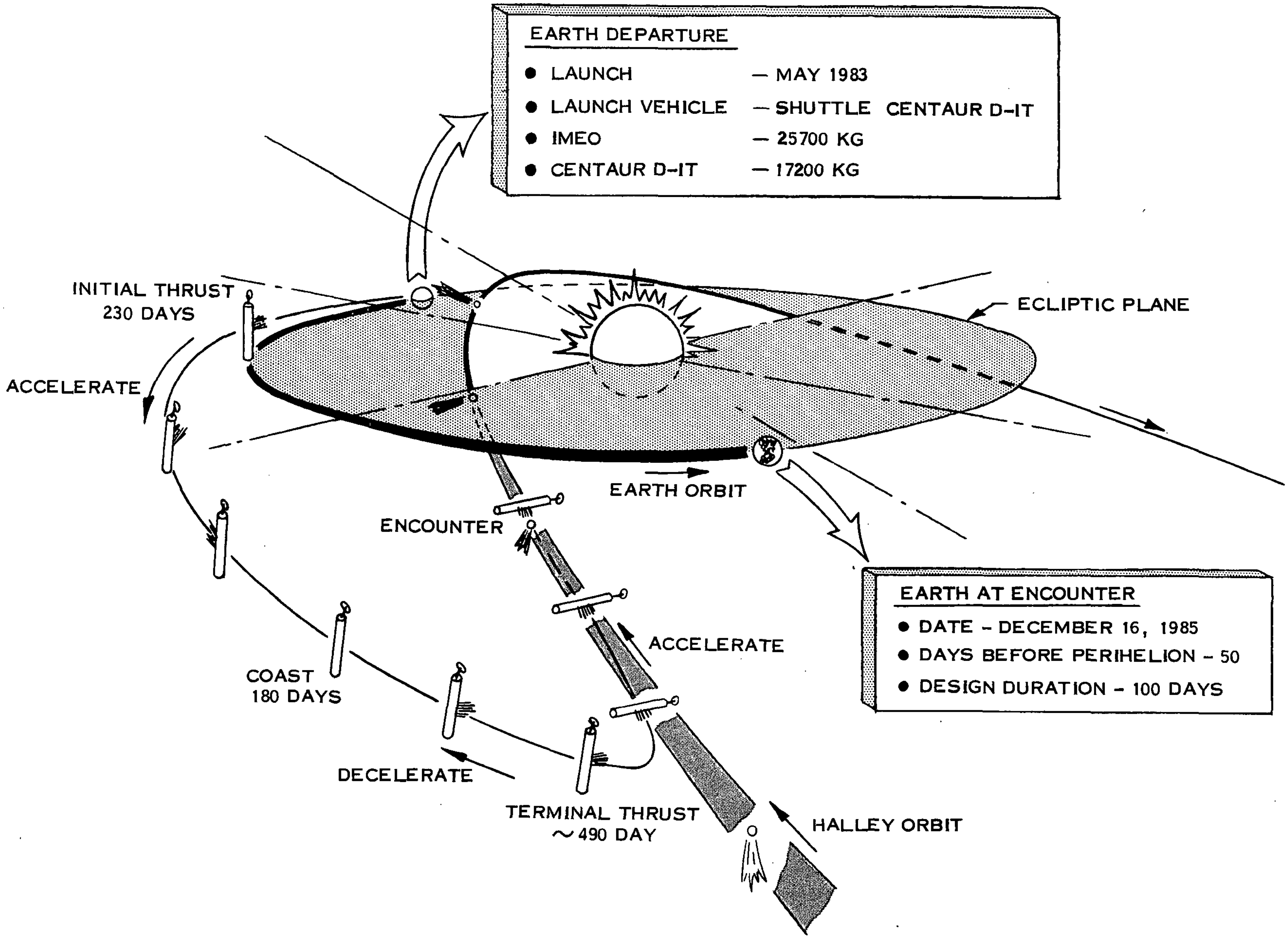


Figure 4-4. Comet Halley Rendezvous Mission (900 Days, 18,000 Full Power Hours)

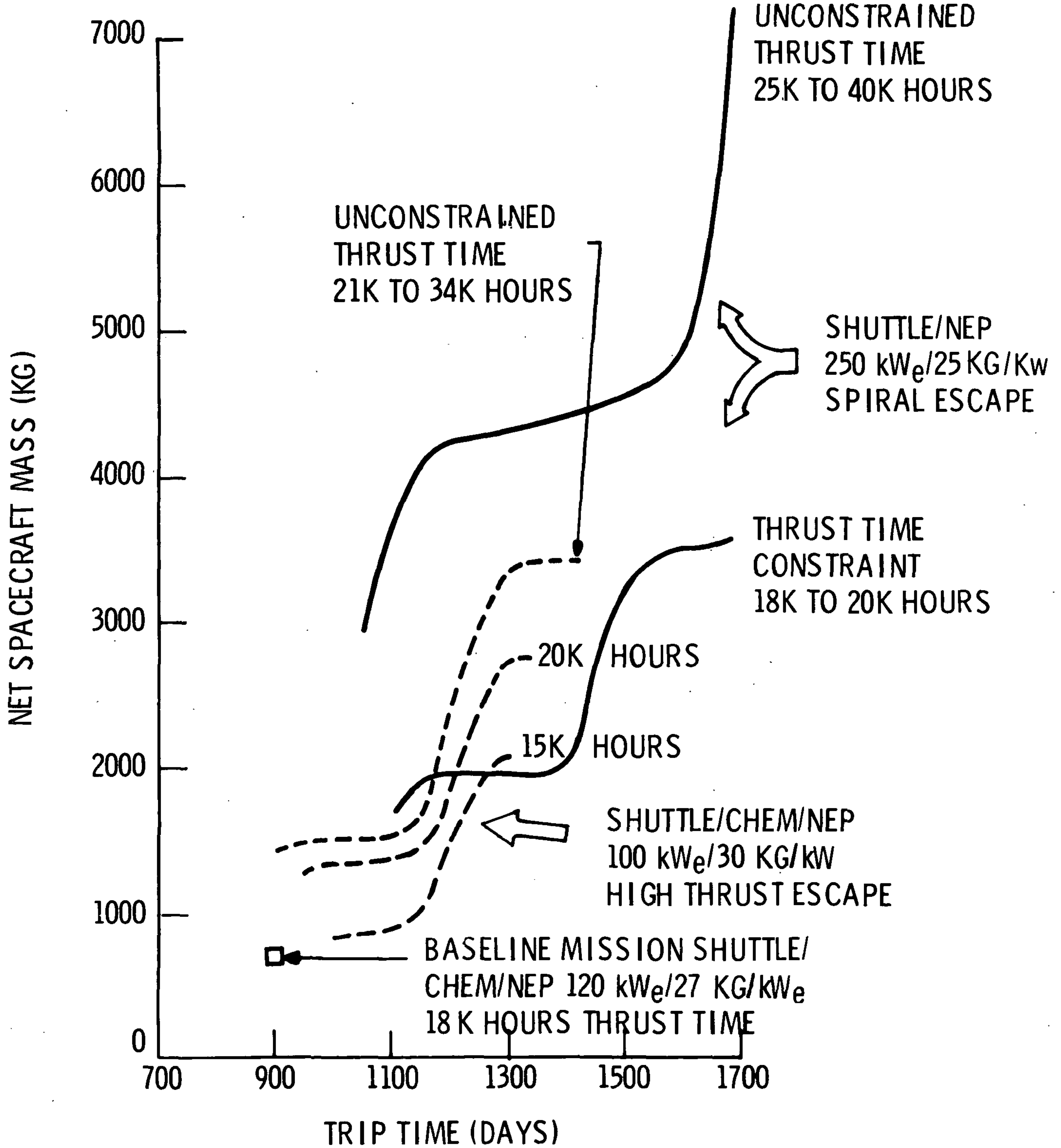


Figure 4-5. Mission Performance of NEP Stage Comet Halley Rendezvous 50 Days Before Perihelion

Comet Halley mission cannot be performed by identified NEP systems if the burn time is constrained to 10,000 hours. The step effect shown on all curves reflects a complete orbit around the sun for the NEP Stage before rendezvous. The payload is increased, but trip times can approach six years.

The baseline Jupiter orbiter mission (illustrated in Figure 4-6) required 14,000 hours of propulsion time, corresponding to a trip time of 900 days. The Centaur D-1T provides a hyperbolic excess velocity of 2.9 km/sec during Earth escape. Of the 900 day trip time, 158 days is used to effect spacecraft descent to a circular orbit of 5.9 Jupiter radii. Since the NEP Stage descends in a slow, nearly circular spiral trajectory, scientific observations can be made throughout the descent through the outer Jovian atmosphere, as well as from the terminal orbit.

Figure 4-7 presents Net spacecraft mass, capture time, and propulsion time as a function of trip time for the Jupiter orbiter mission at 5.9 radii, the orbit of the Jovian moon T_o. The baseline 900 day mission is indicated in the figure.

4.1.4 KEY CONCLUSIONS - MISSION ANALYSIS

The most significant conclusion obtained from the interplanetary mission analysis is the practicability of a multi-mission NEP Stage. This spacecraft is capable of performing not only both baseline interplanetary missions, the Comet Halley rendezvous and the tight Jupiter orbiter, but a large family of outer planet exploration missions as well.

The Shuttle/Centaur D-1T launch vehicle provides superior mission performance relative to the Titan/Centaur family, except for the Titan IIIL4/Centaur. For the outer planet missions, the trip time and propulsion time are not overly sensitive to increases in propulsion system specific mass increases, which may be expected to occur during the propulsion system development program. Of course, such increases must be minimized.

Specific impulse requirements do not exceed 5000 seconds, which should simplify the development of the main power conditioning since the output voltage will not exceed about 3000 Vdc.

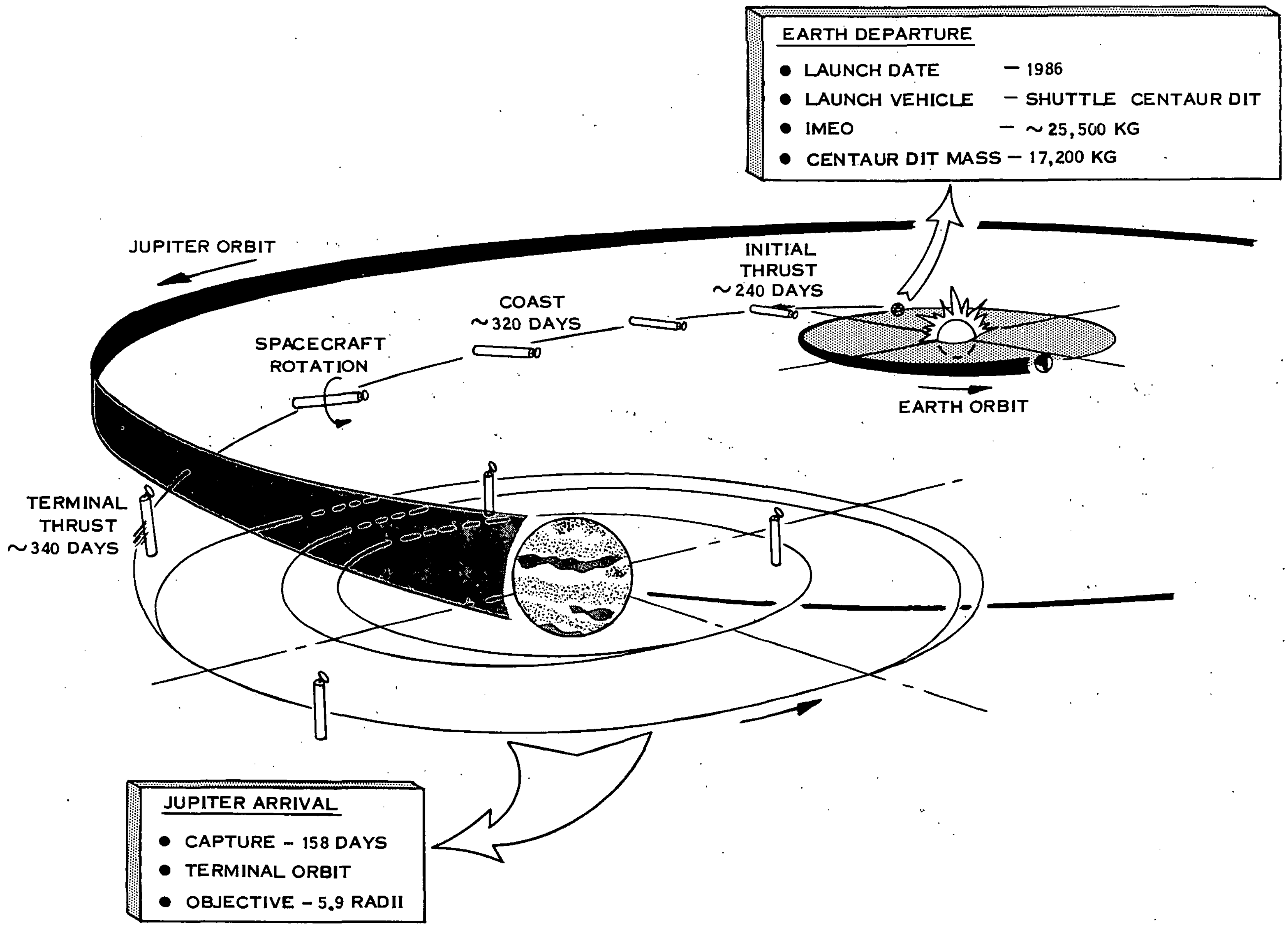


Figure 4-6. Jupiter Orbiter Mission (900 Days, 14,000 Full Power Hours)

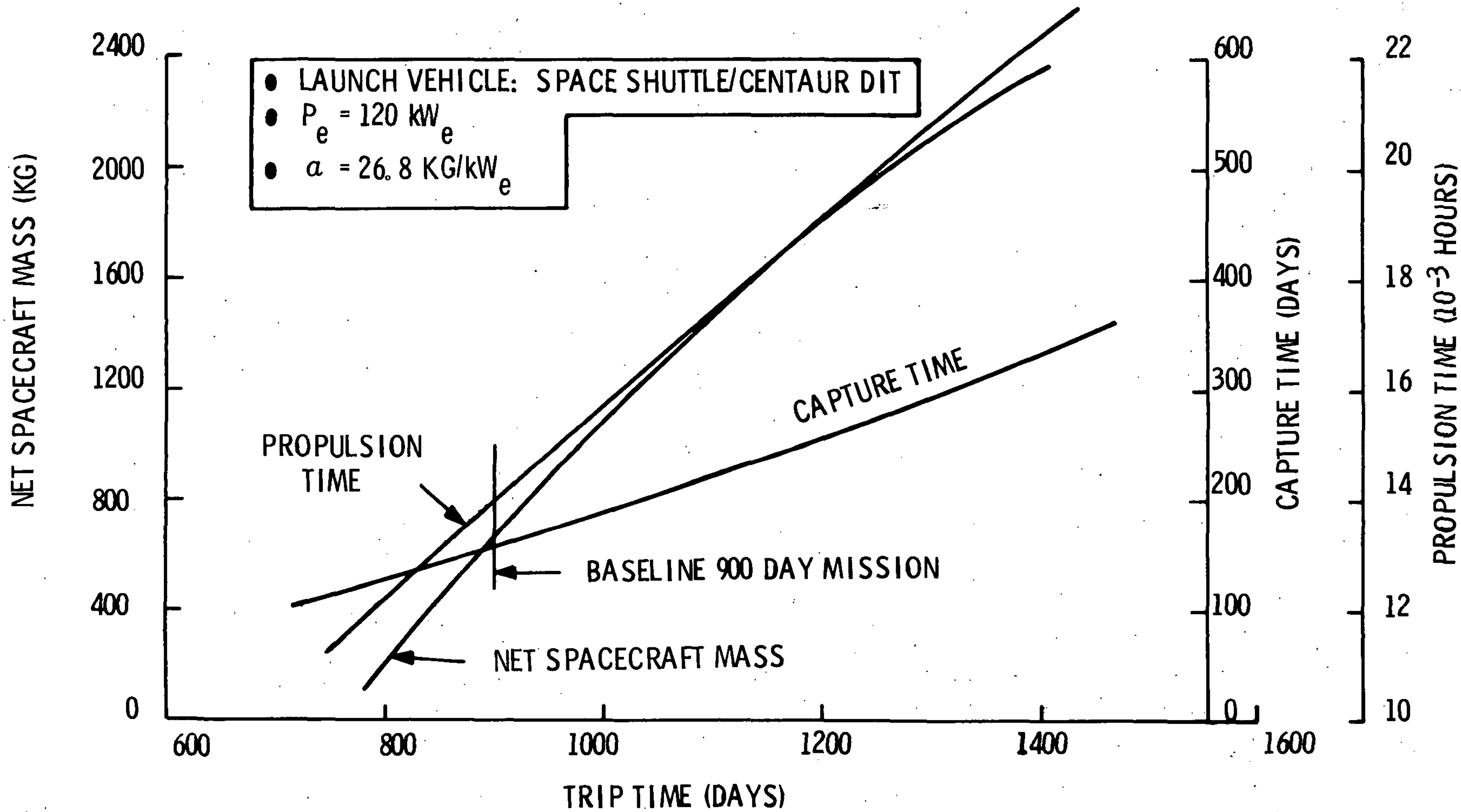


Figure 4-7. Mission Performance of NEP Stage for Mission to 5.9 Jupiter Radii

Propulsion system power levels in excess of that required for minimum energy missions effectively decrease trip time and propulsion time.

4.2 GEOCENTRIC MISSIONS

This section summarizes the mission analysis effort directed toward geocentric earth orbit nuclear electric propulsion applications. The example baseline geocentric orbit mission profile is presented and briefly discussed (see Section 6 for details) and mission performance and payload capability evaluated. Alternate mission profiles are presented and the effects on the baseline geocentric orbit mission of varying certain NEP system and mission related parameters are discussed. The data presented is qualitative in nature and is based on a constant NEP Stage specific mass of 35 kg/kWe.

4.2.1 EXAMPLE BASELINE MISSIONS

The example baseline NEP Stage mission selected for geocentric applications is the transportation of operational payloads to and from synchronous equatorial earth orbit. The mission profile for this baseline geocentric orbit mission is depicted in Figure 4-8. The NEP Stage is Shuttle launched to low earth orbit with a Propellant Logistics Depot (PLD) which stores enough mercury propellant, hydrazine for the reaction control subsystem, and other consumables for the 20,000 hour NEP Stage operational lifetime. The NEP Stage with PLD attached spirals out to a 14,800 by 35,800 km intermediate parking orbit (15 degree orbital inclination) from where it can conduct approximately ten round trip missions to geosynchronous orbit. The Shuttle/Chemical Tug conducts round trip flights to the intermediate orbit to deliver new synchronous orbit payloads to the NEP Stage and to return spent payloads to earth for possible refurbishment.

The 14,800 by 35,800 km intermediate parking orbit is selected because it is above the Van Allen radiation belt and is identical with that selected for Solar Electric Propulsion (SEP). This mission profile minimizes the exposure of the synchronous orbit payload to Van Allen radiation (because of the minimum transfer time obtainable with the Chemical Tug), reduces the trip time to synchronous orbit (relative to an all NEP mission mode), and increases the payload capability to synchronous orbit (relative to that obtainable with the Chemical Tug alone).

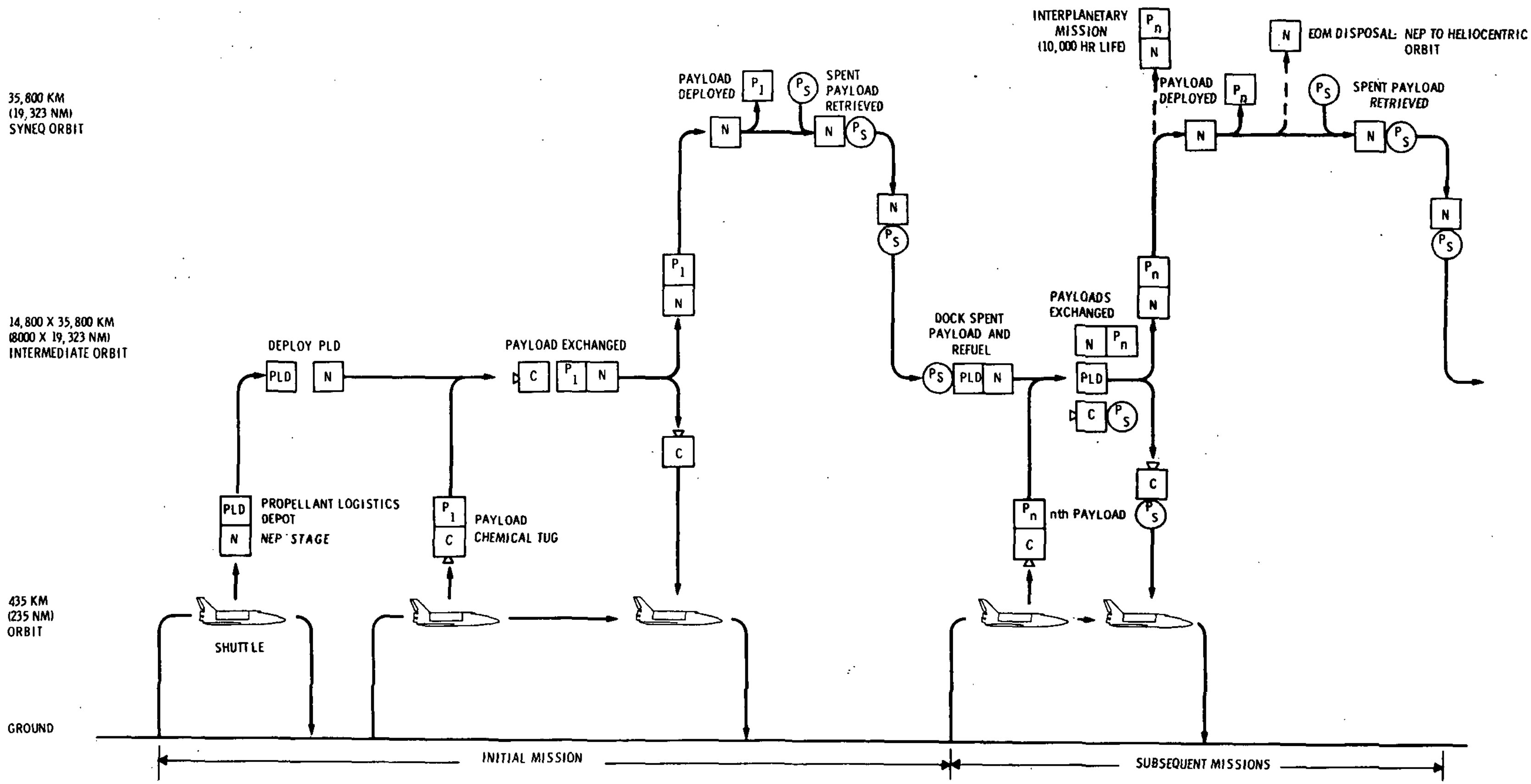


Figure 4-8. Baseline NEP Stage Geosynchronous Orbit Mission Profile

After the NEP Stage has completed its 20,000 full power hour life, it inserts itself into a heliocentric orbit for disposal. The option also exists for the NEP Stage to perform an interplanetary missions after completing up to 10,000 full power hours in geocentric orbit.

4.2.2 GEOCENTRIC MISSION PERFORMANCE

Trip time and payload capability* for the baseline NEP Stage geocentric orbit mission are presented in Figure 4-9. The initial spiral ascent of the NEP + PLD from low earth orbit to the selected intermediate parking orbit will take approximately 140 to 160 days. From the 15 degree inclined intermediate orbit, NEP round trip times are less than 100 days with a maximum payload capability of ~ 7600 kg for equal payload up and back, and 8100 kg for payload placement only. The two round trip (RT) curves of Figure 4-9 are based on the Chemical Tug bringing up the required mercury propellant for the subsequent NEP mission in addition to the operational payload to be delivered to synchronous orbit. Since the baseline geocentric orbit mission includes NEP Stage in-orbit refueling by means of the PLD, the maximum payload mass (Figure 4-9) delivered to the intermediate parking orbit by the Chemical Tug can be increased (with a corresponding slight increase in trip time) by the mass of mercury propellant which would be off-loaded. Therefore, for the baseline mission, maximum payload capabilities of approximately 8600 to 8700 kg are possible with trip times of about 100 days for equal payload up and back, and about 65 days for payload placement only.

Table 4-10 shows the total payload capability of the NEP Stage operating in the baseline geocentric orbit mission mode with a total full power lifetime of 20,000 hours. As the "average" payload mass to be placed in geosynchronous orbit increases, the flight time per mission also increases. This results in fewer possible round-trips over the 20,000 full power hour life of the NEP Stage; however, the total payload mass placed in geosynchronous orbit over the 20,000 hour lifetime increases.

---

*Geocentric mission analysis data presented in this report was performed by NASA-Ames as a part of the Advanced Propulsion Concepts Committee study - Phase I - 1972 (Reference 4-4).

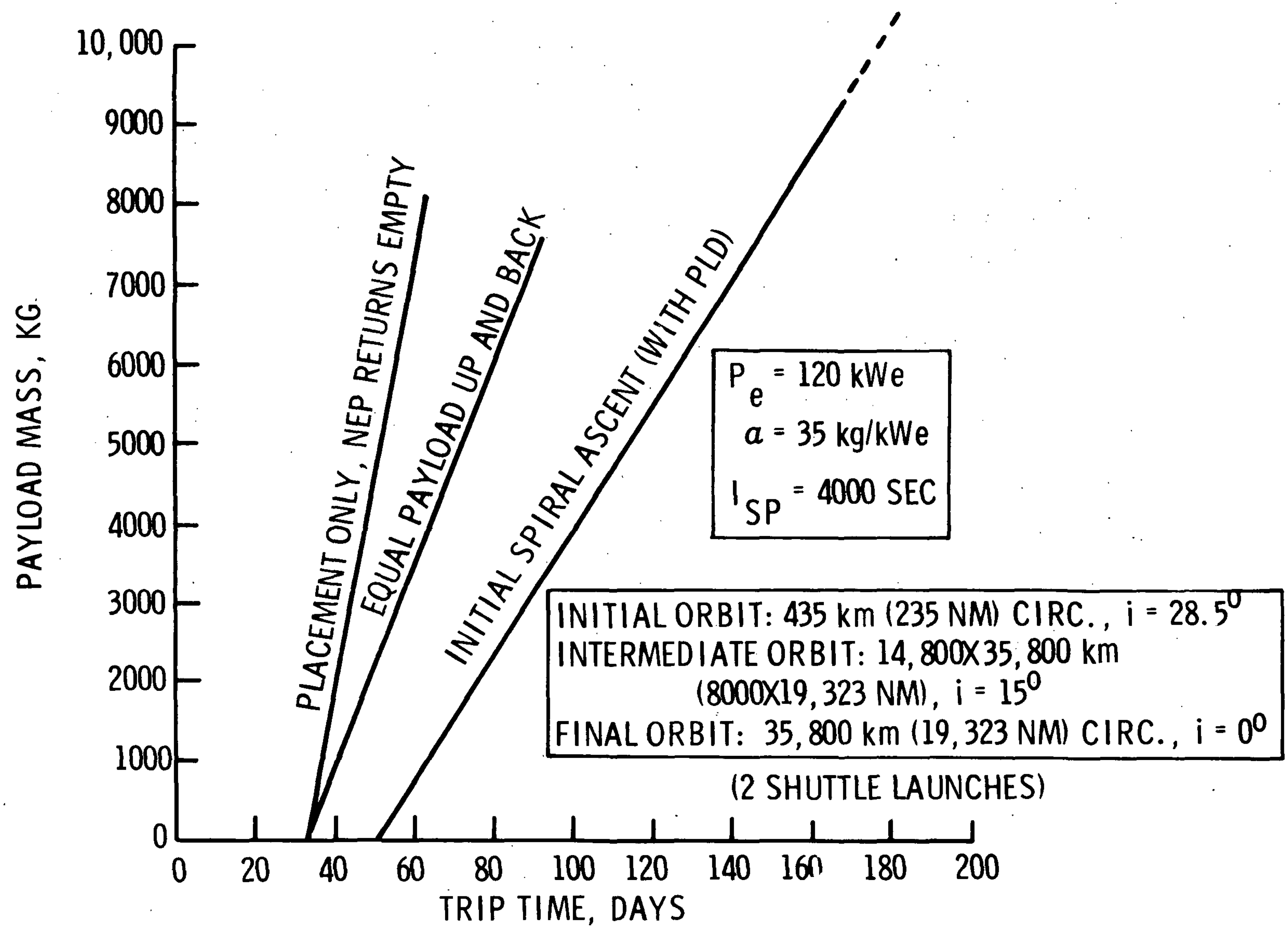


Figure 4-9. Baseline Geosynchronous Orbit Mission NEP Tub/Chemical Tug Performance

Table 4-10. NEP Chemical Tug Payload to Geosynchronous Orbit

Equal Payload Up and Back

Payload Mass	Flight Time (R. T.)	No. of R. T.'s	Max. Payload
100 kg	40 days	17.3	17,300 kg
3000	56	12.3	36,900
5000	72	9.6	48,000
7000	88	7.8	54,500

Placement Only, NEP Stage Returns Empty

Payload Mass	Flight Time (R. T.)	No. of R. T.'s	Max. Payload
100 kg	36 days	19.2	19,200 kg
3000	44	15.7	47,000
5000	51	13.5	67,500
700	59	11.7	82,000

- 120 kWe
- α = 35 kg/kWe
- I_{SP} = 400 sec
- 14,800 x 35,800 km (8000 x 19,323 nm) Intermediate Orbit, $i = 15^{o}$
- 20,000 Full Power Hour Life

- NEP represents Excess Payload Capability for Small Single Payloads
- Multiple Payloads Per Mission Should be Evaluated for NEP

A single NEP Stage, operating in the baseline geocentric mission mode, can deliver (and return) up to 58,000 kg to geosynchronous orbit during its nominal operating life of about seven round trip missions (based on 20,000 full power hour design life). The nominal operating life can be extended to ten round trip missions if the total up and down payload mass is reduced to 46,000 kg. A 30,000 hour design life for a 1986 IOC could result in up to 89,000 kg delivered to geosynchronous orbit during an operating life of approximately 10 round trip missions. If the mission mode is placement only and return empty, the NEP Tug can transport up to 90,000 kg to geosynchronous orbit in about 10 round trip missions.

Because of the large payload capability of the NEP Stage, delivery of multiple payloads to synchronous orbit during one transfer mission should be considered.

4.2.3 ALTERNATE EXAMPLE MISSIONS

In addition to the example baseline NEP Stage geosynchronous orbit mission, several alternate missions have been identified. Two mission modes for the fast delivery (~ 6 hours) of payloads to synchronous equatorial orbit are depicted in Figure 4-10.

The first mission mode involves the Chemical Tug transporting a synchronous orbit payload and NEP Stage to geosynchronous orbit. The payload is deployed and the NEP Stage is used to return the spent Chemical Tug to the intermediate parking orbit for return to the Shuttle by the next Chemical Tug sortie.

Another fast delivery mission mode again involves the payload being transported to geosynchronous orbit by the Chemical Tug. After the payload has been deployed, an NEP Stage (which has been waiting in geosynchronous orbit since deploying a payload of its own) rendezvouses with the Chemical Tug, docks, and returns the spent Chemical Tug to the intermediate orbit for return to the Shuttle by the next Chemical Tug sortie.

In both of these mission modes, the option exists for the NEP Stage to return the spent Chemical Tug to low earth orbit directly rather than to the intermediate parking orbit.

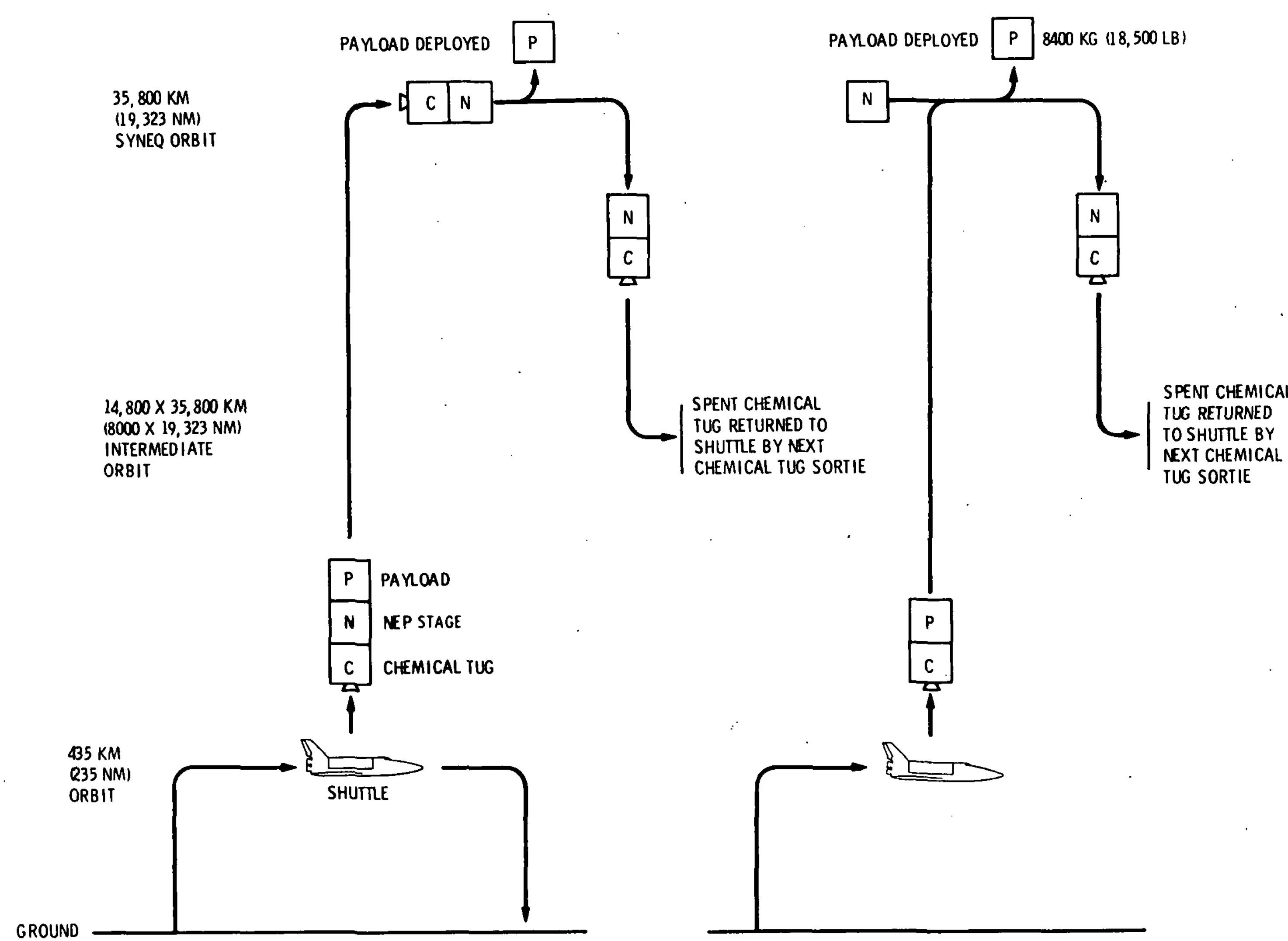


Figure 4-10. Alternate NEP Stage Geosynchronous Orbit Mission Modes (Fast Delivery ~6 Hours)

The all-NEP mission represents another NEP Stage geosynchronous orbit mission alternative. In this mission mode, the NEP Stage with payload spirals out to geosynchronous orbit and back with no chemical assist. This mode of operation is depicted in Figure 4-11.

4.2.4 OTHER GEOCENTRIC MISSION CONSIDERATIONS

Van Allen radiation protection will be required for the power conditioning electronics and certain avionics subsystem electronics. Depending on the spiral out time through the radiation belts, the synchronous orbit payload may also require electron and proton radiation protection.

The effects of specific impulse and inclination of the intermediate parking orbit on mission trip time and payload mass are shown in Figure 4-12. Lowering the specific impulse from 4000 to 3000 sec is seen to result in a mission trip time reduction of approximately 20 percent with an associated small loss in payload capability. The loss in payload capability results primarily from the increased mercury propellant inventory required for the lower specific impulse system.

Maximum payload capability peaks slightly around a 10 to 15 degree intermediate orbit inclination. In addition, a significant reduction in mission trip time can be obtained by placing a portion of the plane change requirement on the Chemical Tug. In the baseline NEP Tug mission, the inclination of the intermediate parking orbit is 15 degrees. This requires the Chemical Tug to perform a 13.5 degree plane change while attaining the 14,300 km by 35,800 km (800 by 19,323 nm) intermediate orbit. If the Chemical Tug performs none of the required 28.5 degree plane change, the NEP Stage round trip time increases approximately 80 percent. If the Chemical Tug were to perform the entire 28.5 degree plane change, NEP mission round trip time could be reduced from 95 days to 10 days. This mission mode would, however, be accompanied with a small loss in maximum payload capability.

The impact of increased power level on mission performance is illustrated in Figure 4-13. In the baseline geocentric mission, a 240 kWe NEP Stage will reduce the spiral round trip time from 93 days to approximately 65 days (~ 30 percent reduction); however, the maximum

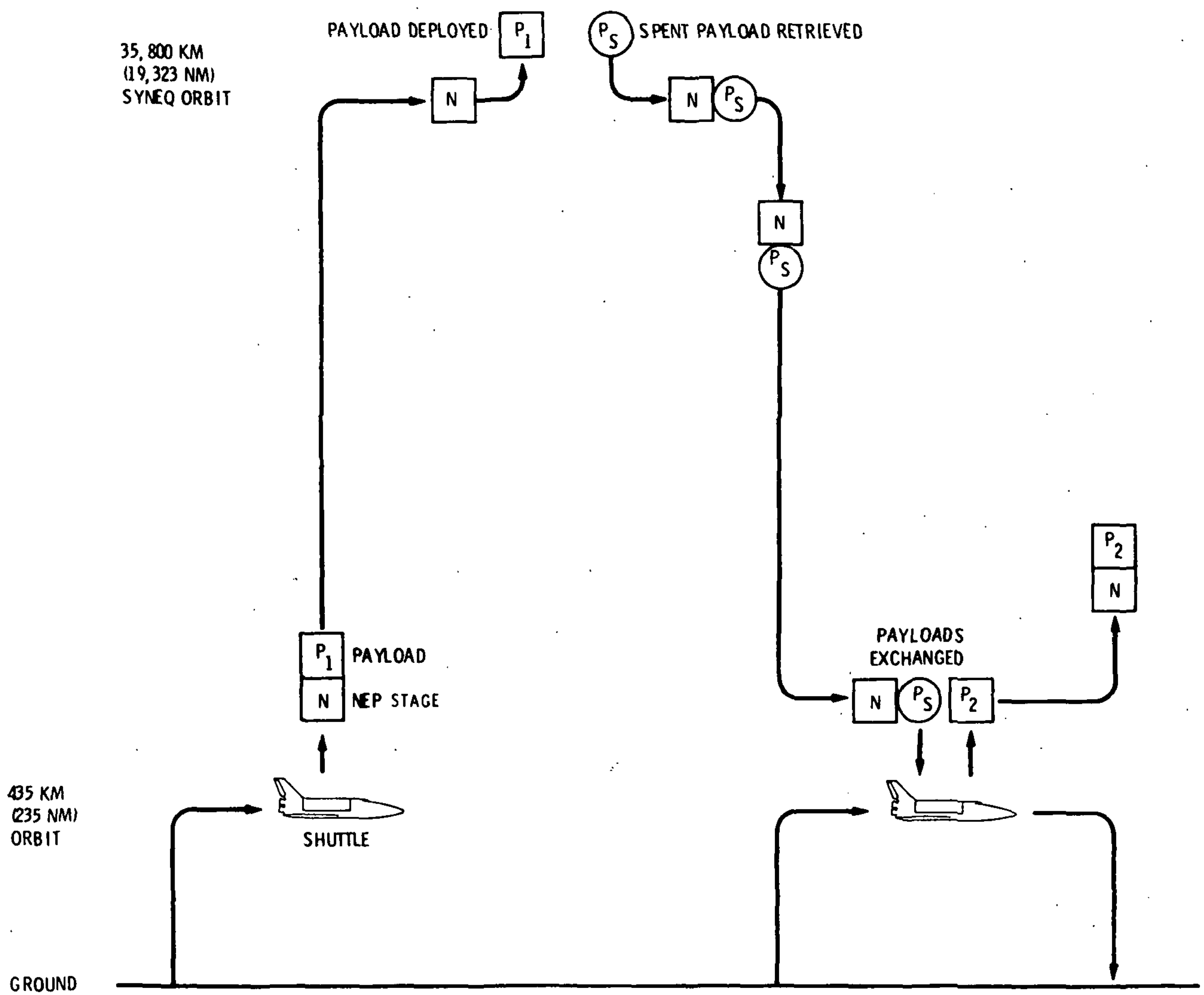


Figure 4-11. Alternate NEP Stage Geosynchronous Orbit Mission Modes (All NEP)

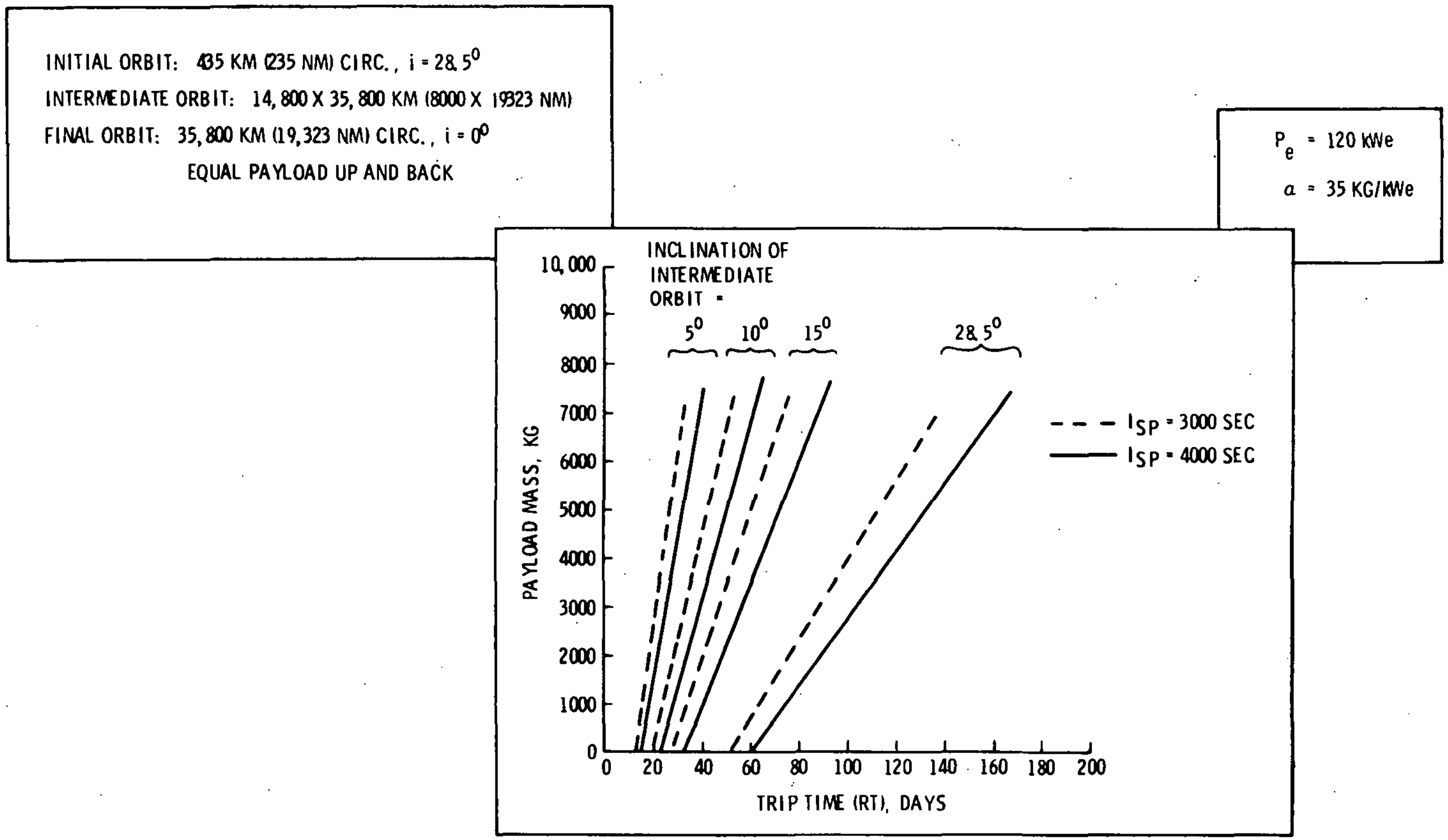


Figure 4-12. Effect of Specific Impulse and Intermediate Orbit Inclination on Mission Performance

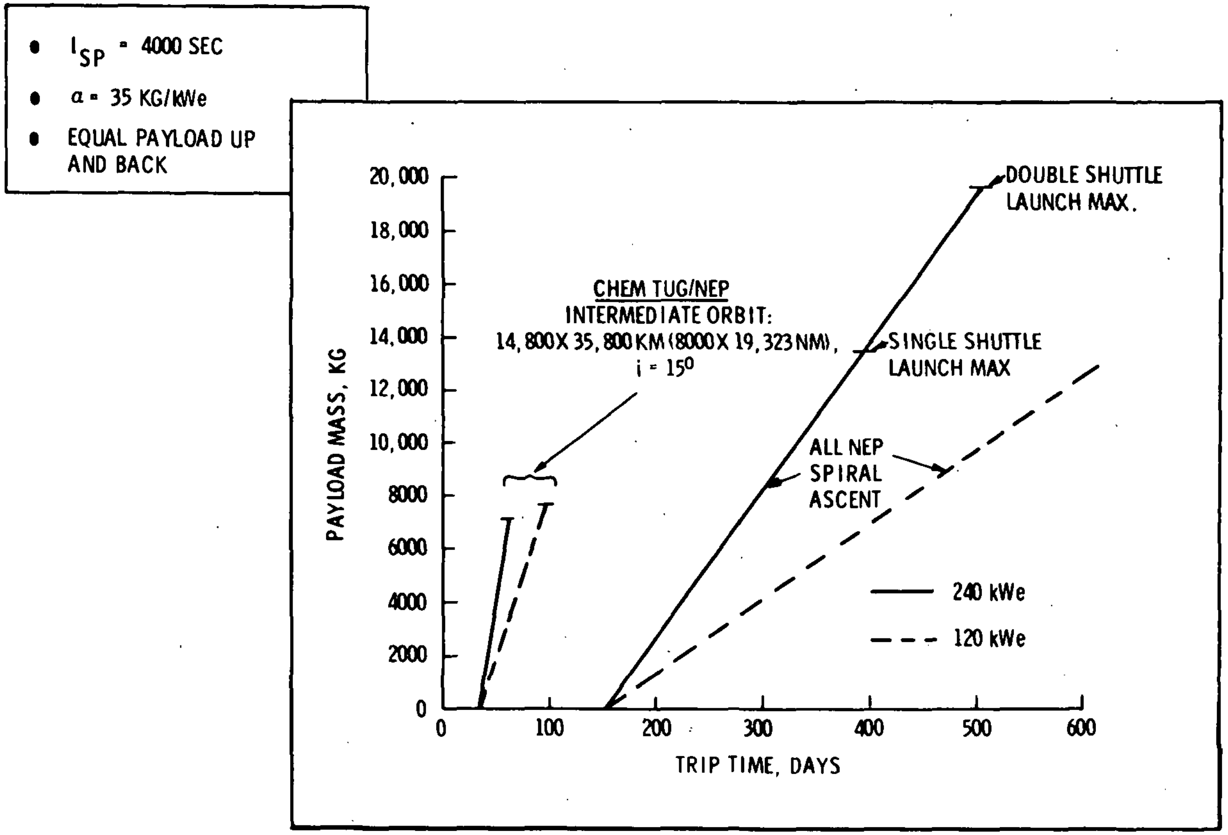


Figure 4-13. Effect of Power Level on Mission Performance

payload capability is also reduced from 8600 to 8300 kg (due in part to the increased mercury propellant requirements for the 240 kWe system).

The impact of higher power level is most noticeable in the mission mode which involves no Chemical Tug assist. In this mode, the NEP Stage travels between low earth orbit and synchronous equatorial with no intermediate orbit. At maximum payload capability with two Shuttle launches, the round trip flight time is reduced from ~900 days for the 120 kWe Stage to ~500 days for the 240 kWe Stage with only a 5 percent reduction in payload capability. Therefore, higher power levels (relative to 120 kWe) are required to make the all NEP mission mode attractive. The optimum power level for this application may in fact be greater than 240 kWe.

Figure 4-14 shows the performance of the NEP Stage in the all NEP mission mode (spiral ascent from 435 km low earth orbit to synchronous equatorial orbit) as a function of power level and specific impulse. Superimposed on these curves is a plot of maximum Shuttle Payload capability to the low Earth orbit. The "knee" of the maximum Shuttle payload capability curves is slightly above 300 kWe. Therefore, Figure 4-14 indicates that the optimum power level for geocentric orbit missions is around 400 kWe.

Final optimization of the mission profile for the NEP Stage operation in geocentric orbit remains to be performed. The 14,800 km by 35,800 km intermediate orbit was selected to reduce the impact of the Van Allen radiation on the synchronous orbit payload (Chemical Tug provides quick trip time through the belt) and to permit a direct performance comparison with the Solar Electric Propulsion (SEP) mission mode. Another intermediate orbit may result in some improvement in mission performance. An intermediate orbit of 14,800 km circular was briefly examined. NEP trip time estimates appeared to be considerably longer, with a slightly greater maximum payload capability.

The results of the mission analysis performed to date (i.e., trip time and maximum payload capability) are not expected to change significantly as a result of subsequent trajectory optimization.

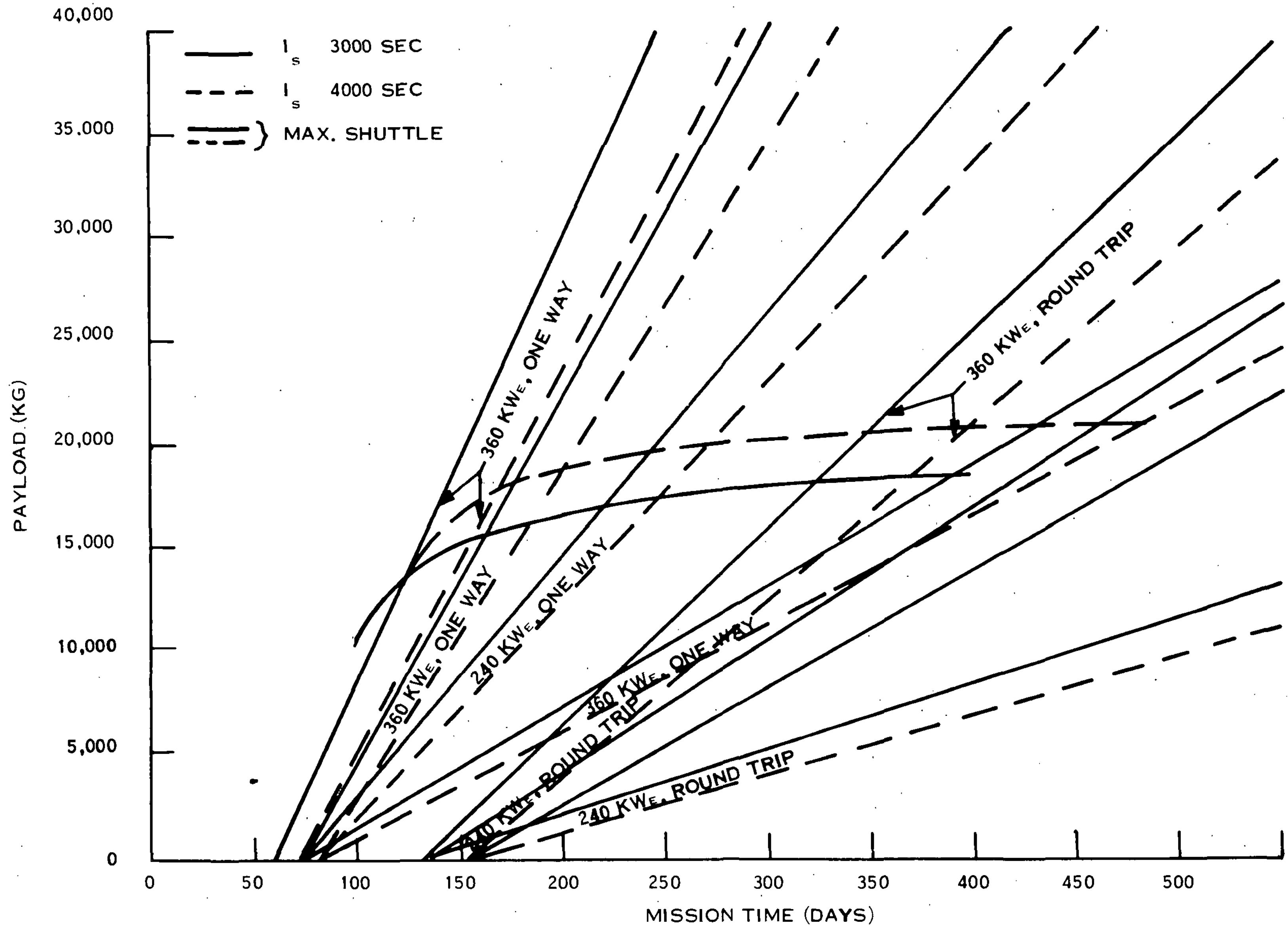


Figure 4-14. NEP Stage Performance to Synchronous Equatorial Orbit (All NEP Spiral Ascent from 235 nm Orbit)

SECTION 5

LAUNCH VEHICLES AND INTEGRATION

The baseline launch vehicle selected for interplanetary mission applications is the Space Shuttle/Centaur D-1T; geocentric orbit missions will use the Space Shuttle or the Space Shuttle/Chemical Tug. This section presents characteristics of these launch systems as well as NEP Stage/launch vehicle integration concepts.

Independent of the type of mission, the Space Shuttle is the baseline launch vehicle used to transport the NEP Stage (and a kick-stage if necessary) from the earth's surface to low earth orbit (435 km). The following ground rules apply to the definition of the Space Shuttle as used in this study:

1. The basic definition of the Space Shuttle will be as presented in the Shuttle RFP issued by MSC until such time as the Shuttle user's guide is re-issued.

2. Shuttle design performance is 29,450 kg gross payload mass to 435 km circular orbit independent of inclination between 28.5 degrees and 50 degrees. Orbit Maneuvering Subsystem (OMS) propellants are utilized to achieve this.

3. Payload handling equipment in the Shuttle orbiter bay for all Shuttle system third stages will be similar, if not common. The tilt cradle or pallet is assumed to be the baseline concept. Cradle and cradle errection hardware, and spacecraft safety monitoring equipment constitute "Shuttle payload support weight."

4. Shuttle orbiter contains no Air Breathing Engine System (ABES).

5. The Shuttle third stage will be ready for orbit launch no earlier than 2.85 hours following Shuttle lift-off.

6. No propellant is shared between the Shuttle orbiter and any of the candidate third stages.

7. Geometric constraints on the orbiter payload are 18.3 m free length by 4.58 m free diameter, as shown in Figure 5-1.

8. The Shuttle orbiter constrains its payload CG to lie within the limits shown in Figure 5-2.

9. The loads imparted to the Shuttle cargo are defined in Table 5-1.

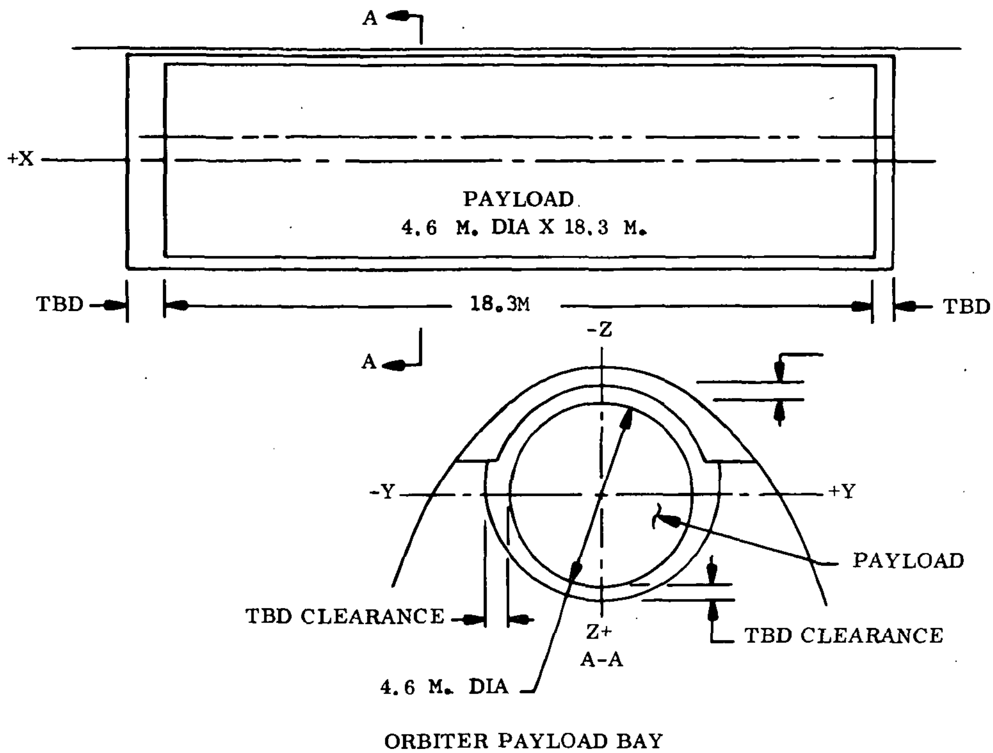


Figure 5-1. Shuttle Orbiter Cargo Bay Envelope

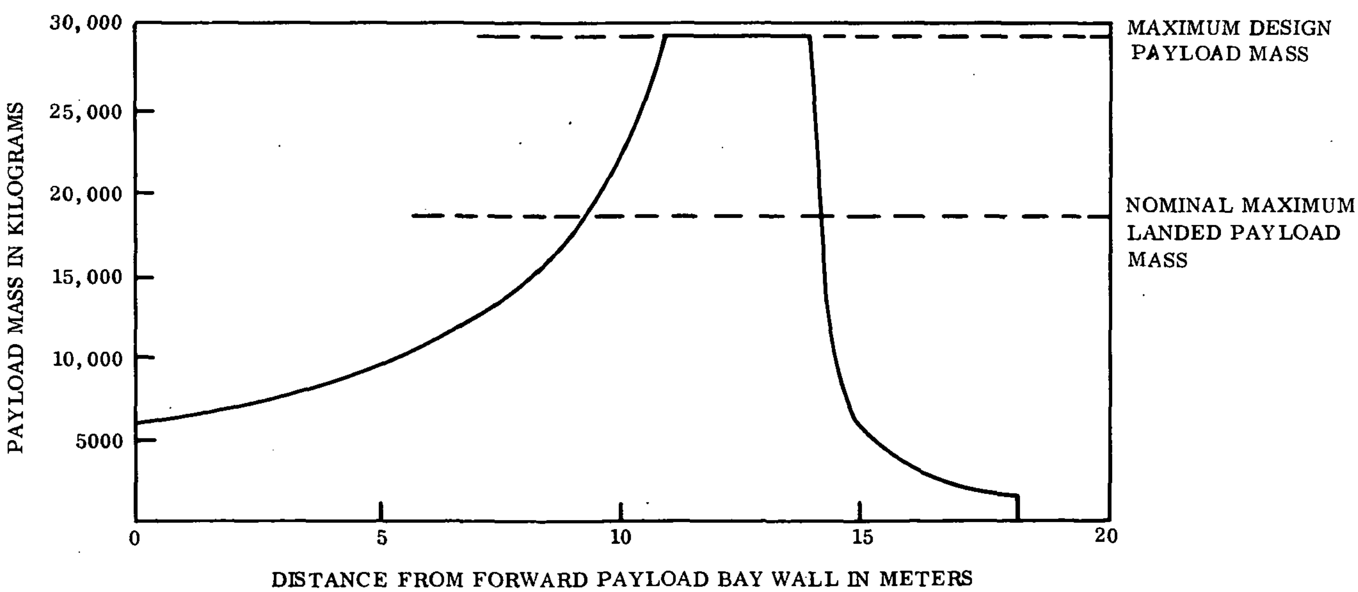


Figure 5-2. Payload Longitudinal Center-of-Gravity Limits

Table 5-1. Shuttle Payload Load Factors

Condition	X(g)	Y(g)	Z(g)
Launch	1.4 ±1.6	1.0	1.0
High-Q Booster Thurst	1.9	±1.0	0.8 ±0.2
End Boost (Booster Thrust)[a]	3 ±0.3	± 0.6	±0.6
End Burn (Orbiter Thrust)	3 ± 0.3	± 0.5	±0.5
Orbiter Entry	- 0.5	± 1.0	-3.0 ±1.0
Orbiter Flyback	-0.5	± 1.0	±1.0 -2.5 ±1.0
Landing	-1.3	±0.5	-2.7 ±0.5

a. Excludes booster-orbiter separation loads which are TBD.

5.1 INTERPLANETARY MISSIONS

Present Shuttle designs can deliver payloads up to 29,450 kg to a 435 km circular orbit. The payload configuration must conform to the baseline Shuttle orbiter cargo bay geometry of 18.3 m long by 4.6 m diameter, because it is most unlikely that a special orbiter would be built to accommodate nuclear electric propulsion spacecraft. In the Shuttle-Centaur launch mode, the Centaur becomes part of the orbiter payload. This further constrains the allowable NEP Stage launch mode configuration, to that area within the fixed orbiter payload bay which is not occupied by the Centaur. The Space Shuttle/Centaur launch vehicle characteristics are presented in Figure 5-3.

To provide full multi-mission capability, the NEP Stage must be designed to be Shuttle launched with the 9.15 m long Centaur D-1T kick stage. Figure 5-4 shows the NEP Stage weight penalty in terms of reactor radiation shielding if a "short" NEP Stage is configured to be placed end-on-end with the Centaur in the Shuttle cargo bay. The figure indicates that

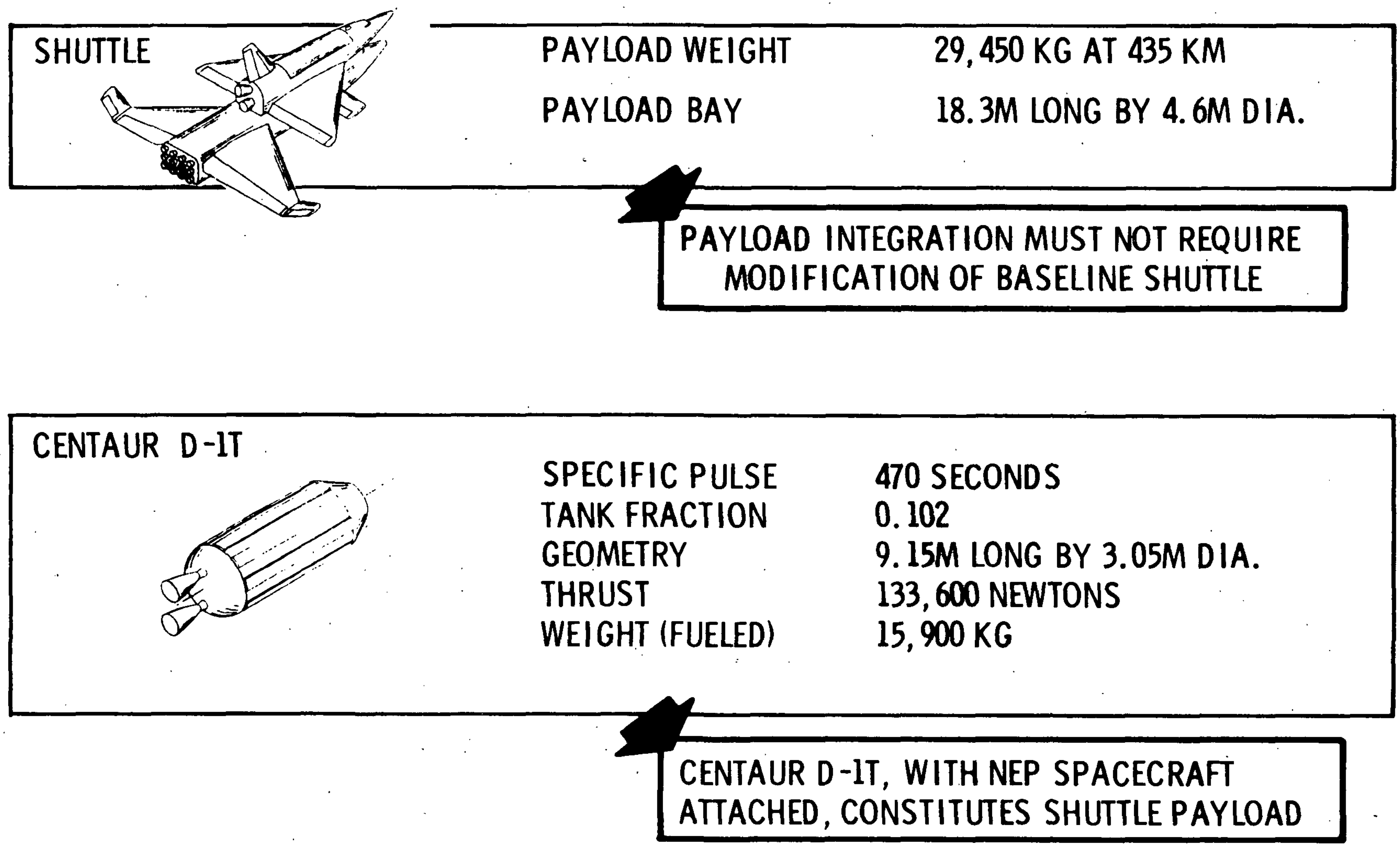


Figure 5-3. Space Shuttle-Centaur Characteristics

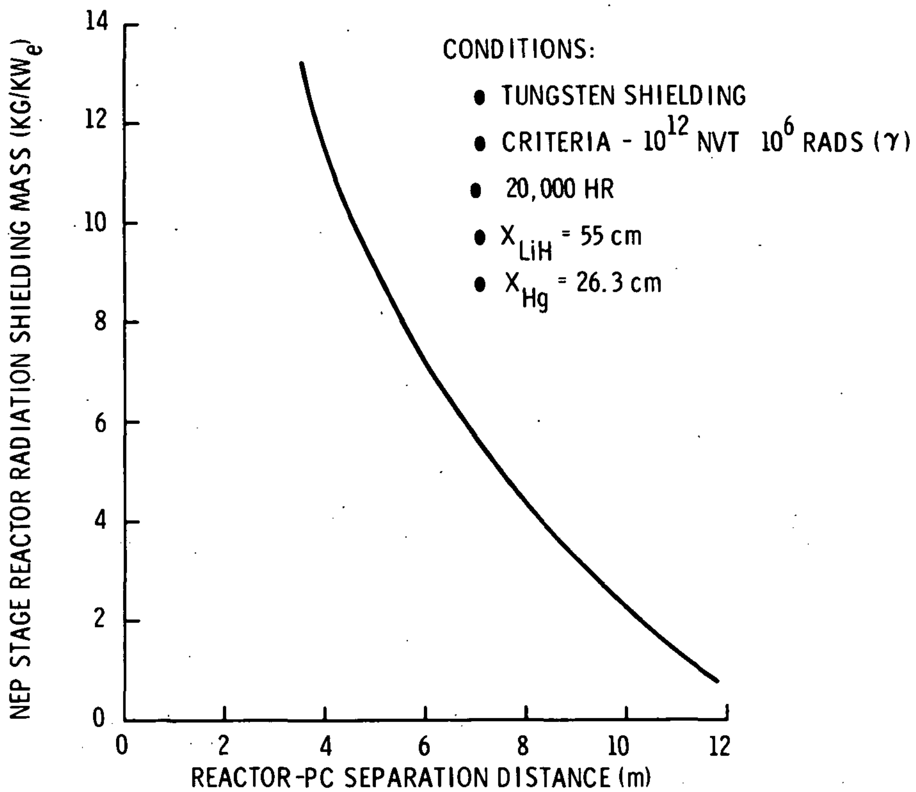


Figure 5-4. Radiation Shielding Weight Penalties

as the NEP Stage is shortened to be packaged end-on-end with the Centaur, the permanent gamma radiation shielding requirements increase significantly. Since this mass contribution is clearly unacceptable, other means of packaging must be developed.

The selected Shuttle packaging configuration for the NEP Stage and Centaur is illustrated in Figure 5-5. Since the forward end of the NEP Stage is basically a hollow cylindrical configuration of 4.6 m diameter, this allows for the 3.05 m diameter Centaur stage to be located inside the NEP Stage for Shuttle packaging. Upon separation of the NEP Stage/Centaur from the Space Shuttle, the mated configuration is oriented to the initial conditions required for injection to earth escape (see Section 6) using the NEP Stage RCS. The Centaur is fired for the high energy earth escape with the forward half of the stage remaining internal to the NEP Stage. This mode of operation presents no real difficulties for the Centaur (Reference 5-1).

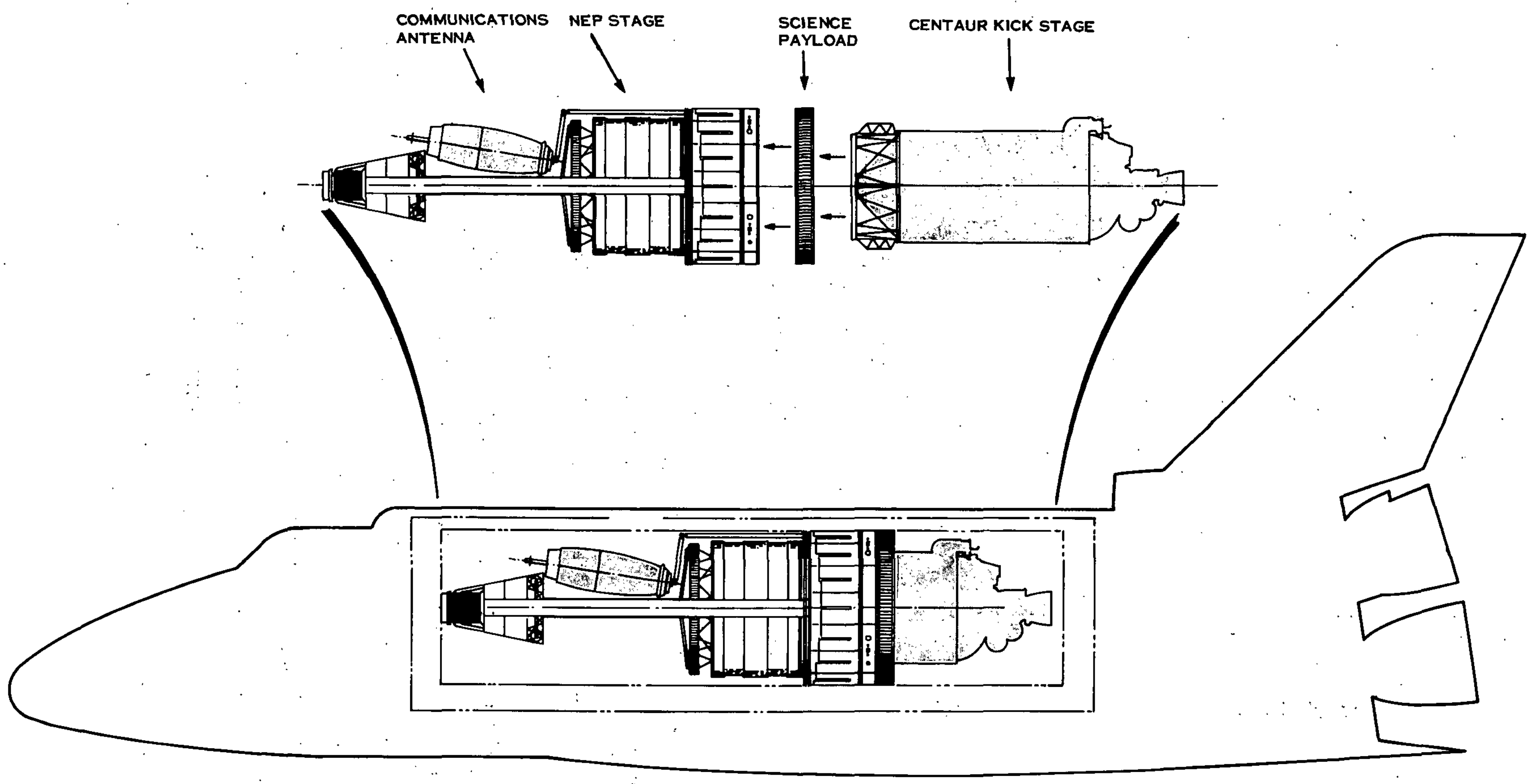


Figure 5-5. NEP Stage/Centaur Shuttle Packaging Concept

After the high energy burn, the Centaur is released from the NEP Stage by firing pyrotechnic separation devices located on the internal support mating ring. This separation scheme could include a spring-activated separation mechanism. The released Centaur is guided out of the NEP Stage by a four-rail guide system to ensure minimum angularity differences during separation. Figure 5-6 shows the NEP Stage/Centaur center-of-gravity location based on the reference interplanetary mission superimposed on the Shuttle payload CG envelope. For the total payload mass including the fully fueled Centaur, it is necessary to locate some portion of the mercury propellant in tanks positioned on the aft bulkhead of the primary radiator to provide an acceptable CG location.

While in the Shuttle cargo bay, the mated NEP Stage/Centaur launch configuration is placed on a transfer module/payload support pallet (a truss-like structure) which interfaces directly with the Shuttle attachment points. This assembly is depicted in Figure 5-7. The NEP Stage/ Centaur configuration is attached to the transfer module at four points:

1. Forward support ring of the reactor/shield configuration.
2. Aft support ring of the primary radiator.
3. Forward support of the science payload ring.
4. Aft section of the Centaur.

The aft section of the transfer module that supports the Centaur state is the same support structure as that of the aft support structure proposed for the Shuttle integration and launch of the Centaur.

An alternate launch vehicle for interplanetary missions is the Titan III. Payload capabilities of the Titan/Transtage and Titan/Centaur families of launch vehicles are presented in Figure 5-8. Typical hyperbolic excess velocities of from 1.0 km/sec to 4.0 km/sec are required for the candidate missions. Typical launch vehicle payload requirements range from 8400 kg to 12,000 kg for the 120 kWe NEP Stage. Therefore, launches with the Titan Centaur family will require availability of the large diameter core Titan, with at least two strap-on solids (Titan IIIL2).

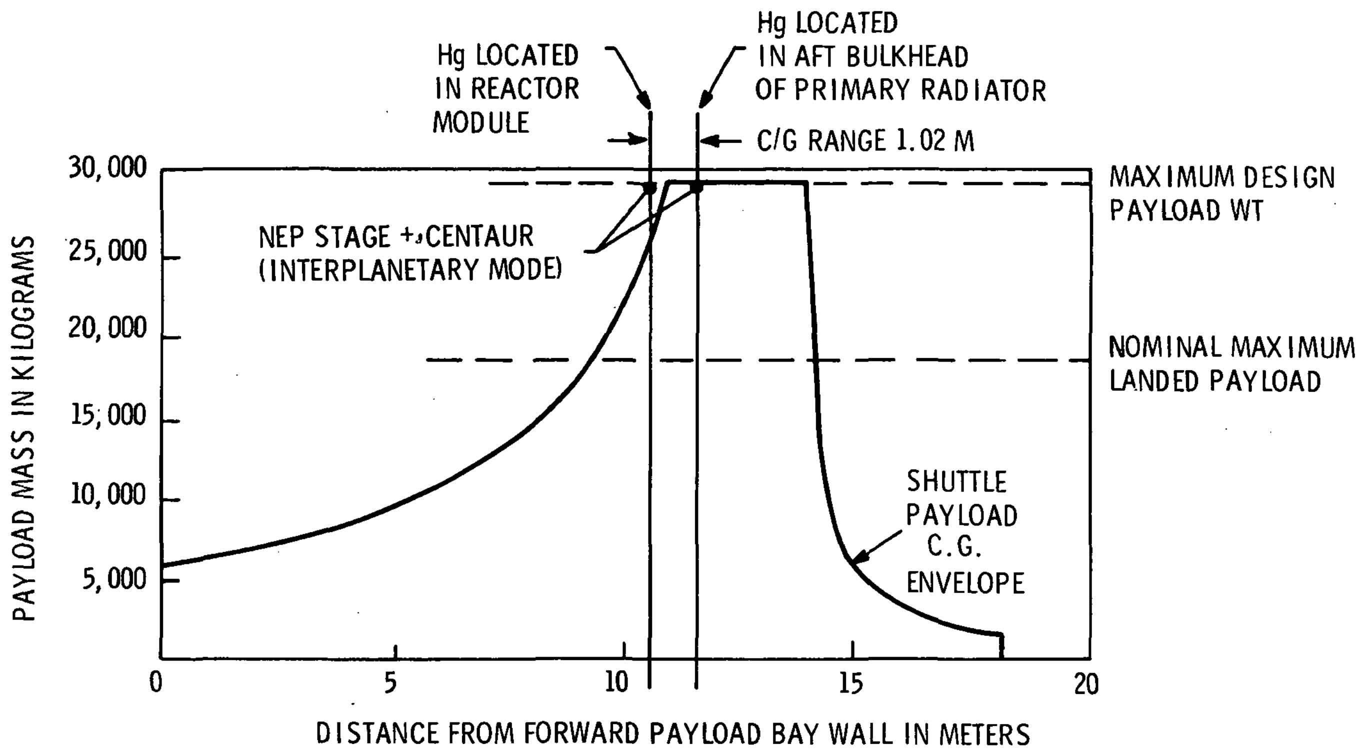


Figure 5-6. Payload Longitudinal Center-of-Gravity Limits for NEP Stage/Centaur

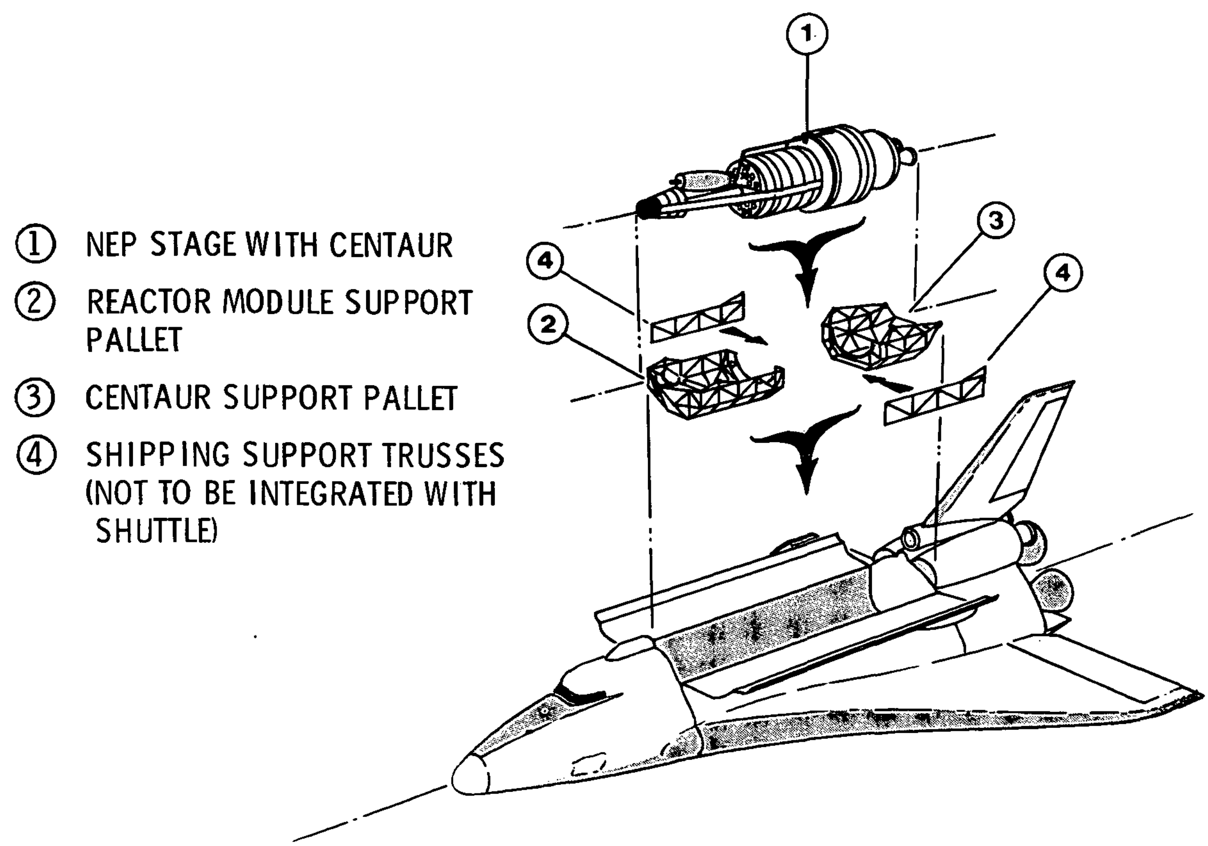


Figure 5-7. NEP Stage/Centaur Shuttle Transfer Module Assembly

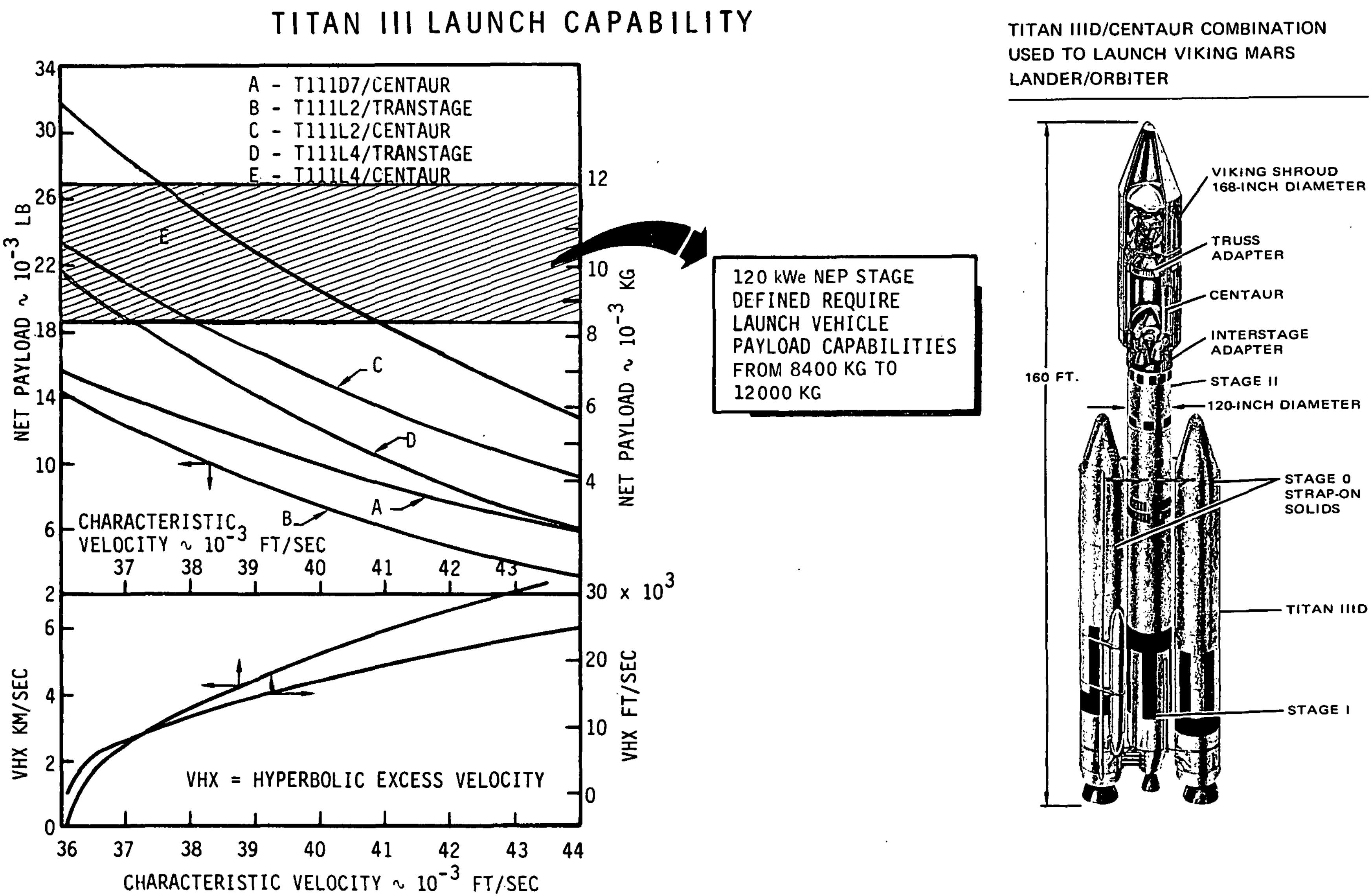


Figure 5-8. Titan III Launch Capability

5.2 GEOCENTRIC MISSIONS

In geocentric mission applications, the NEP Stage will be launched with either a Propellant Logistics Depot (PLD) for in-orbit refueling or a synchronous orbit payload. Figure 5-9 shows the reference NEP Stage packaged in the Shuttle cargo bay with a "payload". The payload is attached to the NEP Stage by the docking structure that extends from the avionics module. The mated NEP Stage/payload configuration is placed on a transfer module that interfaces directly with the Shuttle attach points.

The two preliminary example alternate NEP Stage configurations discussed in Section 3.7.2 were primarily designed for geocentric orbit applications. Figure 5-10 illustrates the Shuttle integration scheme for the 120 kWe side thrust configuration. To be packaged in the Shuttle cargo bay with up to a 7.6 m long payload (or the Centaur/Chemical Tug), the example side thrust NEP Stage is folded as shown. As indicated in the figure, when folded in the Shuttle cargo bay, the 120 kWe side thrust NEP Stage can be transported to low earth orbit while utilizing the maximum Shuttle allowable payload capability without exceeding the Shuttle longitudinal center-of-gravity limits. This is based on payload CG located at the payload volumetric center.

Figure 5-11 shows one Shuttle integration concept for the example 240 kWe end thrust NEP Stage. The example 240 kWe NEP Stage is a deployable configuration which allows it to be packaged in the Shuttle cargo bay. The power conditioning radiator is mounted on a guide-rail system that can be collapsed inside the heat pipe primary radiator. In this configuration, up to a 3.7 m long payload can be accommodated during Shuttle launch.

To meet the Shuttle payload longitudinal CG constraints, the 240 kWe NEP stage must be positioned aft in the cargo bay with payload forward. However, even in this launch configuration, maximum Shuttle utilization is not achieved. This preliminary Shuttle packaging analysis is based on a fixed payload volume and fixed payload CG. If either of these are varied, the curve of NEP Stage/payload center-of-gravity locations will shift to the right or left. In addition, ballast could be used to shift the combined CG within the desired envelope. Further evaluation is required at these higher power levels to maximize Shuttle utilization.

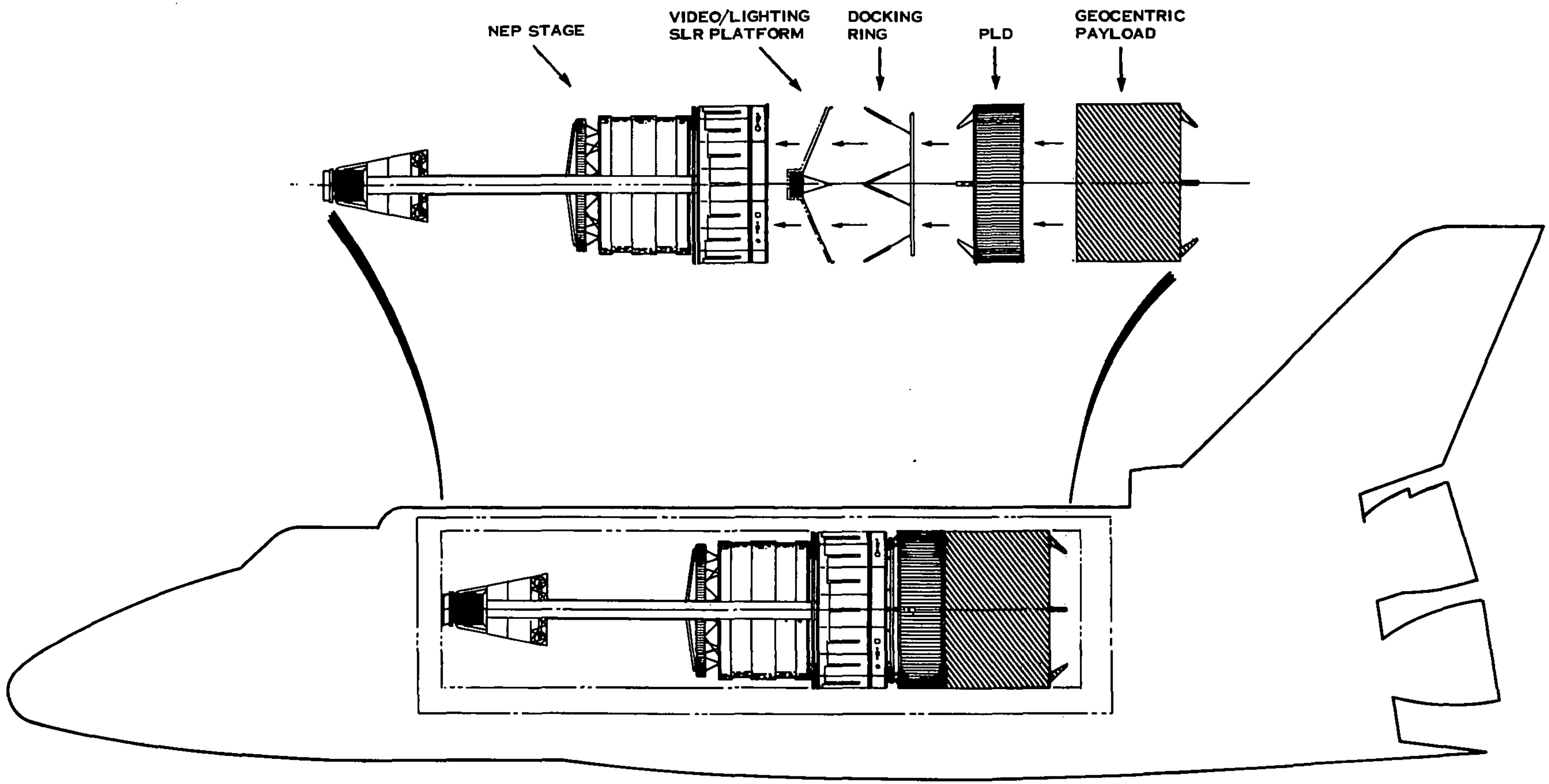


Figure 5-9. NEP Stage in Shuttle Cargo Bay with "Payload" (for Geocentric Mission)

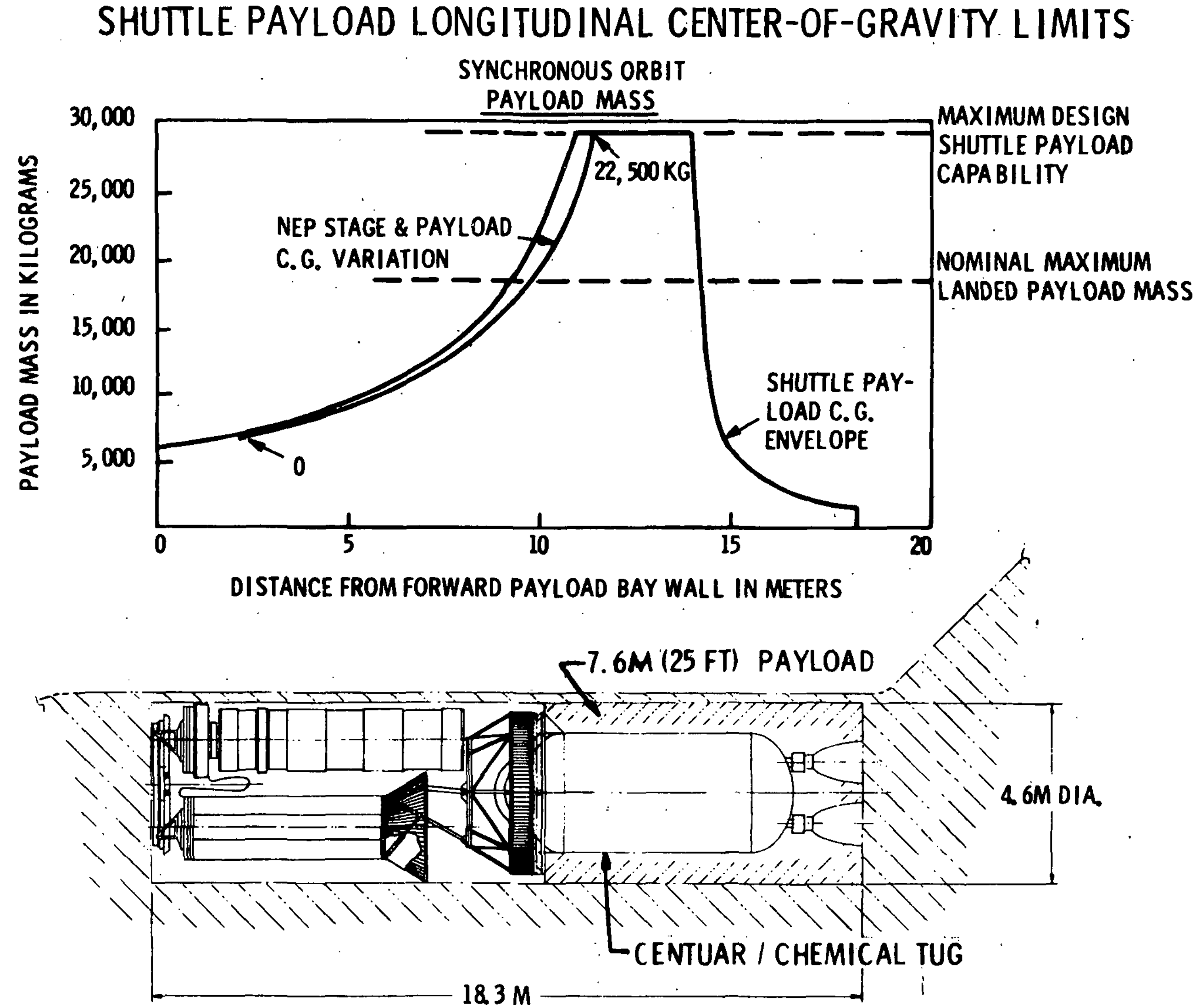


Figure 5-10. Shuttle Integration for 120 kWe NEP Stage Side Thrust Configuration

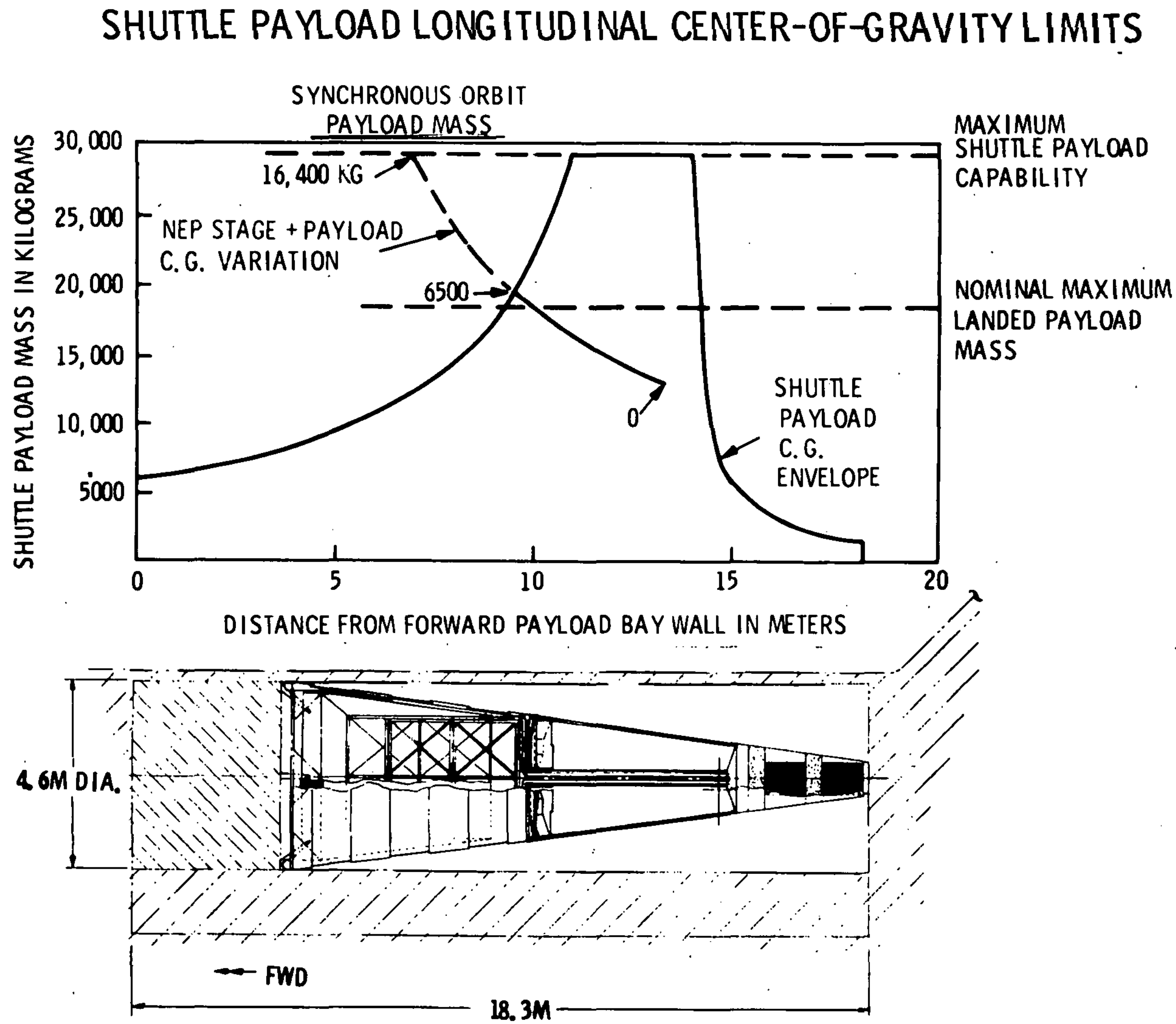


Figure 5-11. Shuttle Integration for 240 kWe End Thrust NEP Stage (Alternative Configuration)

In the selected reference geocentric orbit mission, the Chemical Tug is employed to transport synchronous orbit payloads between the 435 km low earth orbit and the 14,800 km by 35,800 km elliptical parking orbit. The baseline Chemical Tug is shown in Figure 5-12. The performance, packaging and deployment of the Chemical Tug in the Shuttle orbiter is based on the Baseline Tug Definition Document (Reference 5-2). During Shuttle transport, the Chemical Tug is located forward in the cargo bay with its payload aft. Primary structural support of the payload while in the Space Shuttle is from the Chemical Tug/payload structural interface located at the forward end of the Tug. The engine thrust level is 44,480 Newtons (10,000 lb) with a specific impulse of 470 sec. The CG is approximately 4.31 m (170 in.) forward of the nozzle exit plane.

The forward support ring (see Figure 5-13) provides for attachment of the aft end of the Chemical Tug to the Shuttle. This ring incorporates two major fittings for the total axial support of the tug and lateral support in one direction.

The maximum initial mass of the Chemical Tug and payload is 29,450 kg. This represents the injected payload capability of the Shuttle into a 435 km orbit with a due East launch.

The reference Chemical Tug has the capability to deploy and retrieve a 9,050 kg payload at the 14,800 km by 35,800 km intermediate orbit ($i = 28.5^{o}$) and return to the 435 km ($i = 28.5^{o}$) orbit from which it initially departed to allow rendezvous and docking with the Shuttle. This capability is reduced to approximately 8,500 kg if the inclination of the intermediate orbit is reduced to 15 degrees.

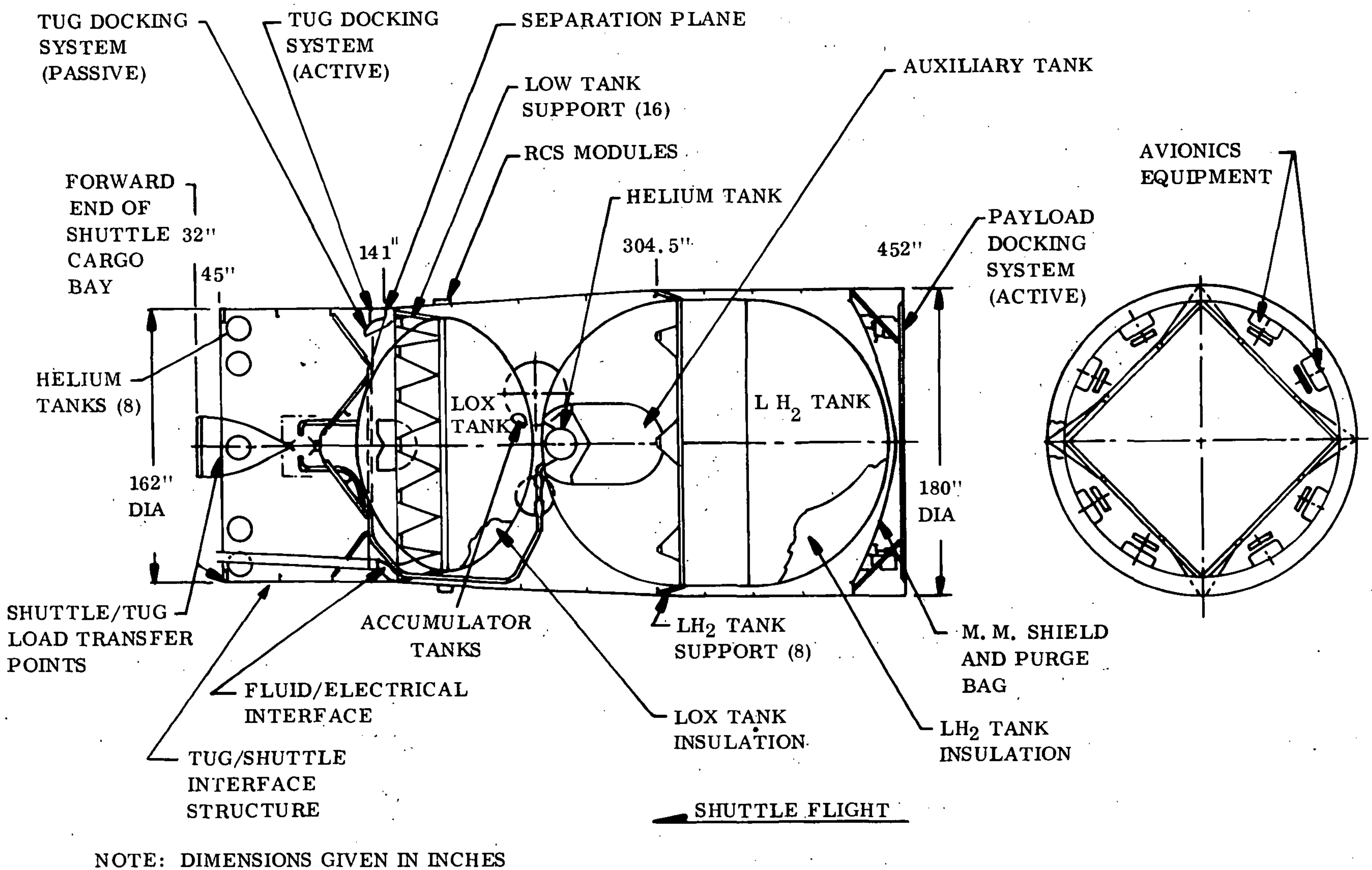
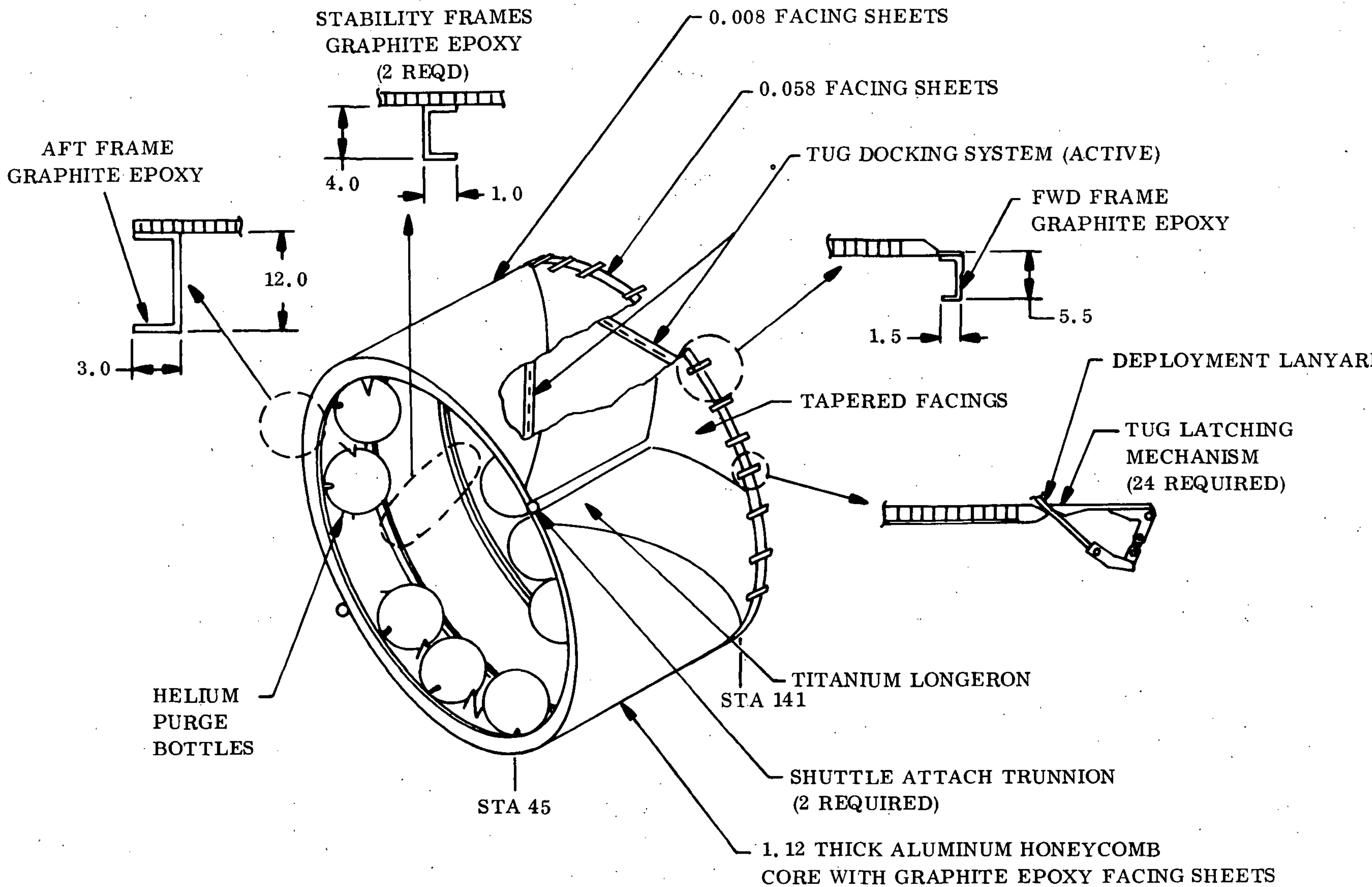


Figure 5-12. Baseline Chemical Tug Configuration



NOTE: DIMENSIONS GIVEN IN INCHES

Figure 5-13. Shuttle/Chemical Tug Adapter

SECTION 6
MISSION OPERATIONS

The objective of the mission operations analysis is to determine the effect of operational activities on the NEP Stage design and development cost. Design constraints include size, mass, configuration, and requirements dictated by safety conditions, fabrication and test procedures. Cost influences are mainly auxiliary or supplementary hardware which are needed on both the NEP Stage and the ground to successfully complete the mission. This section identifies the Ground Support Equipment (GSE) and operational equipment required for the operation of the NEP Stage. Section 7 summarizes the GSE and operational equipment and briefly discusses the key hardware and facilities.

A number of assumptions were made to limit the operations analysis to the most probable course of action. These assumptions are listed in Table 6-1. Also, the Space Shuttle is assumed to be the workhorse launch vehicle in the 1980's.

Table 6-1. Key Assumptions Mission Operations

Common	Space Shuttle Launch
	The power subsystem will be completely assembled, sealed, tested at reactor fabrication site
	Performance of ion engines as an array will be tested at spacecraft assembly facility
	The NEP stage will be completely assembled and tested prior to shipment to launch site
	Operational checks of reactor drums will be performed one at a time with all other units safety locked
	In-space flight operations, with possible exception of navigation function and rendezvous and docking functions will be controlled by an on-board computer
Interplanetary Missions	NEP stage and Centaur packaged together in space shuttle cargo bay
	Reactor startup will not be permitted until after earth escape has been achieved
	Final navigation operations will be guided by an on-board planet/comet detector unit
Geocentric Missions	Baseline mission is the same as that selected initially for Solar Electric Propulsion (SEP)
	Propellant Logistics Depot (PED) required to support NEP Stage
	PLD will have on-board attitude control and tracking capability

In order to maximize the probability of mission success, it is assumed that:

1. The NEP Stage will be fully assembled and tested, as required, prior to shipment to the launch site
2. The ion engines are to be tested integrally with the thrust subsystem
3. The reactor control drums are locked in the shutdown position during all ground activities after the reactor has been assembled and before liftoff
4. The liquid metal loops of the power subsystem are seal welded after filling and testing at the reactor fabrication site.

A preprogrammed, on-board computer is assumed to sequence and control all flight operations, with the possible exception of navigation and rendezvous and docking. A back-up ground control mode will be utilized only if a situation occurs that has not been programmed into the computer or if a computer function(s) fails.

For interplanetary missions, cost considerations and the added complexity of in-orbit assembly decree the joint launch of the NEP Stage and the Centaur kick stage. For safety considerations, reactor operation is prohibited until the vehicle has been successfully propelled out of earth orbit. A planet/comet detector unit is assumed to be included in the interplanetary science payload to direct the final flight corrections leading to planet/comet interception.

The following subsections discuss the NEP mission operations for both interplanetary and geocentric orbit missions. The fabrication and test, prelaunch, and launch phases will be the same for either type of mission.

6.1 INTERPLANETARY MISSIONS

The NEP interplanetary mission has been divided into six phases for purposes of the mission operations analysis. The phase designations and their definitions are:

Phase 1: Fabrication and Test

This phase includes all manufacturing, assembling and testing of NEP Stage components,

subassemblies, subsystems, etc., up to and including the complete vehicle. The phase is completed with shipment of the NEP Stage to the launch site.

Phase 2: Prelaunch

This phase includes all activities at the launch site up to liftoff.

Phase 3: Launch and Earth Orbit

This phase includes all operations from liftoff to separation of the NEP Stage from the Space Shuttle in earth orbit.

Phase 4: Earth Escape and Near Earth Operations

This mission phase begins with chemical propulsion for earth escape and ends with the initiation of electric propulsion by the NEP Stage.

Phase 5: Heliocentric Flight

This phase includes the activities from the beginning of electric propulsion until the approach by the NEP Stage to the target planet/comet.

Phase 6: Planet/Comet Arrival

This phase begins with the arrival of the NEP Stage into the near vicinity of the planet/comet and is terminated by completion of the experimental survey of the target planet/comet.

The mission operations analysis for interplanetary NEP applications is based on a side thrust NEP Stage configuration since this work was performed before the identification of the end thrust NEP Stage configuration. All of the interplanetary mission operations discussed will be the same regardless of the NEP Stage configuration, except for the Fabrication and Test Phase and the Earth Escape and Near Earth Operations Phase.

6.1.1 FABRICATION AND TEST

The fabrication and test operations are divided into major activities which basically deal with the manufacture, assembly and acceptance testing of the main subsystems of the NEP Stage. These major activities, not necessarily in sequential order since many of the subsystems will be fabricated concurrently, are as follows:

1. Power subsystem fabrication and test

2. Power conditioning assembly fabrication and test

3. Thrust bay assembly fabrication

4. Thrust bay assembly performance test

5. Propellant subsystem fabrication and test

6. Thrust subsystem assembly and test

7. Propulsion system assembly and test

8. Avionics subsystem/interplanetary science payload fabrication and test

9. NEP Stage assembly and test

Figure 6-1 presents the flow chart for the Fabrication and Test Phase. This phase will be basically the same for the end thrust NEP Stage; however, some additional difficulties are introduced with the fabrication and test of the end thrust configuration because of the different arrangement of the components with respect to the side thrust configuration.

6.1.1.1 Power Subsystem Fabrication and Test

The power subsystem consists of the thermionic reactor, the main heat rejection radiator, the EM pumps, accumulators and piping comprising the coolant loop between the reactor and radiator, the neutron shield, and miscellaneous structure, control units and instrumentation circuits. Fabrication of these components will be performed at various locations, but the assembly of the power subsystem will be done at the reactor fabrication site to utilize its nuclear handling facilities.

The design of the power subsystem in the side thrust NEP Stage configuration has the reactor and most of the other components, surrounded completely by the main heat rejection radiator. Consequently, the reactor is fabricated and assembled, joined to its structural support, then built up with the addition of the EM pumps, piping, etc., of the heat rejection loop. Copper electrical power leads are attached to the TFE junctures and control actuators and instrumentation circuits are added to appropriate coolant loop piping.

Because of the overriding importance of power subsystem operation to mission success,

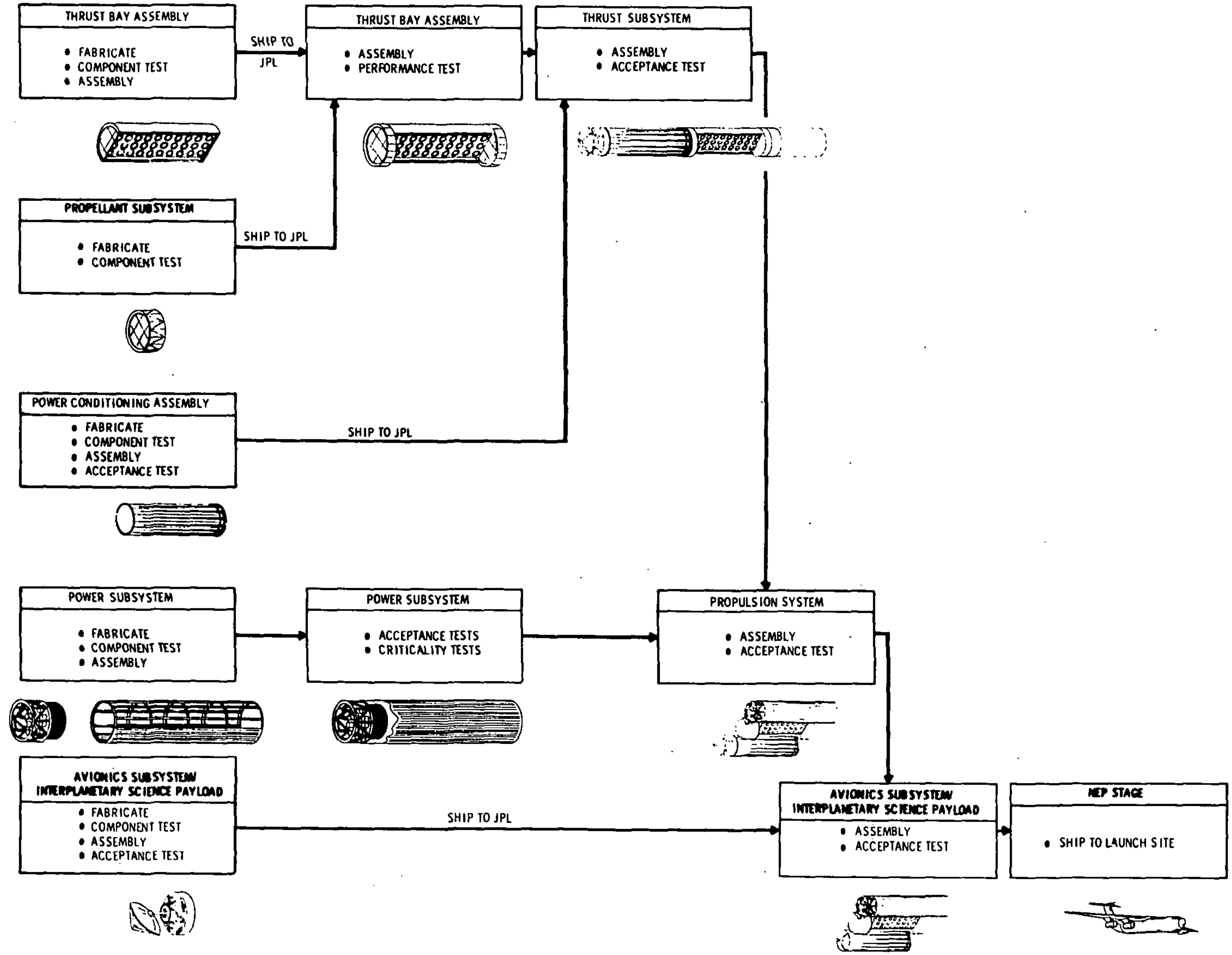


Figure 6-1. Phase 1: Fabrication and Test

stringent tests are performed during fabrication and assembly. During fabrication of the reactor, each TFE is performance checked before placement in the reactor, and all welds which are in contact with NaK coolant are checked. All the other heat rejection loop weldments are similarly checked.

In the assembly of the reactor, the control drums are tested one at a time for freedom of operation, then safety wired or locked in the shutdown position. With the addition of actuators to the control drum shafts, a similar check of operational freedom is made.

After complete assembly of the power subsystem, a gas leak check is performed on the coolant loop circuit at a "hot" temperature. Then the loop at room temperature is filled with NaK, pressurized, checked for coolant leakage, then cold trapped by circulation through ground equipment until purity standards are achieved. Cold flow tests and calibrations of the coolant system using system pumps are performed.

Hot tests of control drum, EM pumps, cesium heater and reactor instrumentation operation are conducted. A simulator which duplicates reactor control signals is attached to the power subsystem, and simulated reactor startup and shutdown tests are performed. The shield assembly is then joined to the power subsystem assembly.

The power subsystem is installed in a nuclear facility and zero power, cold criticality tests are performed to determine critical drum positions, worth, etc., and compared with expected values. If facilities are available, a similar set of tests is performed at an advanced temperature level.

The tested reactor is packaged in a special shipping container with attached environmental control facilities and shipped to JPL.

6.1.1.2 Power Conditioning Assembly Fabrication and Test

The power conditioning assembly fabrication begins with the construction of the radiator framework. The various electrical cable subassemblies, such as the power subsystem

control and instrumentation cables, the ion engine control and instrumentation cables, the ion engine high voltage power cables, etc., which traverse the length of the power conditioning assembly, are attached to the inner sections of the framework. Each of the twenty-four main power conditioning modules is assembled on its individual radiator panel and the panel then joined to the radiator framework. The special ion engine PC modules are assembled on appropriate panels and the panels attached to the frame. The low voltage power cables for each of the main power conditioning modules are strung and attached along the outer surface of the radiator.

Each of the main and special ion engine power conditioning modules is tested by applying design electrical power voltage of each of the low voltage power cables and measuring the output from the corresponding output lead. The tested assembly is then packaged in a special shipping container and transported to JPL.

6.1.1.3 Thrust Bay Fabrication

Fabrication of the thrust bay assembly commences with the construction of the structural frame and the subsequent installation of the translator bed and mechanism. Pretested ion engines, set in the gimbal mechanism mounts, are then installed on the translator bed. Operation of the translator mechanisms and each of the gimbals are checked. Propellant feed lines and control valves are connected to each ion engine and leak tested. Electrical power and instrumentation cables are similarly connected and continuity checked to each engine.

Control and instrumentation cabling for the reactor, pumps, etc., of the power subsystem, which cross the length of the thrust bay, are installed and checked for continuity. Aluminum bus bars for reactor output power are attached with insulated connectors to the rear surface of the thrust bay framework. A prepackaged auxiliary power supply with associated control units and circuitry is then mounted behind the ion engine platform and tested for proper operation. The disposable launch support beams are fixed in position across the open end of the thrust bay.

The final operation is the packaging of the assembly in a suitable container, and shipment to JPL.

6.1.1.4 Propellant Subsystem Fabrication and Test

Each propellant tank assembly is welded to design, then leak checked for external leaks and for internal leaks between the pressurized gas and mercury compartments. Suitable coating and shipment of the tank assemblies terminate this subphase.

6.1.1.5 Performance Test of Thrust Bay

Performance testing of the ion engine array will be performed at the spacecraft assembly site, assumed to be JPL. A receiving inspection on the thrust bay assembly including an operation check of the translator and gimballing mechanisms is made as well as a visual inspection of the propellant tanks. The propellant tanks are installed on the thrust bay and connected to the propellant feed lines. A leak check is performed on the propellant system, mercury is added to the tanks and a check of transfer operations between tanks is made, if appropriate.

The performance test requires vacuum facilities of sufficient size to hold the thrust bay and special test equipment. A high voltage electrical power source of $\sim$110 kWe and a 40-volt power source of $\approx$10 kWe is needed for input to the ion engines. In addition, a simulator that duplicates the ion engine control and switching functions of the avionics subsystem as well as the control of the translator and gimbal mechanisms is required. The test will consist of the following:

1. Startup, shutdown and restart operations on individual engines
2. Simultaneous full power operations on all primary engines
3. Simulation of ion engine failure and automatic switching to redundant engines
4. Operation and control of translator and gimbal mechanisms with full power operation of engines

After a successful test and drainage of mercury from the propellant subsystem, the thrust bay is either packaged for temporary storage or transported directly to the spacecraft assembly area.

6.1.1.6 Thrust Subsystem Assembly and Test

The thrust subsystem consists of the power conditioning assembly and the thrust bay assembly with the latter containing the propellant subsystem. A receiving inspection which includes a performance check of each power conditioning module is conducted on the power conditioning assembly. If the thrust bay assembly has been stored for any length of time, it also is checked. The two assemblies are joined structurally and the corresponding power and control cables connected.

The acceptance test for the thrust subsystem requires special equipment of the following capability:

1. A low voltage electrical power source to duplicate the output power of the reactor
2. An electrical load bank as a substitute for the ion engines
3. A simulator which duplicates the thrust subsystem control functions of the avionics subsystem

The acceptance test consists of providing design electrical power to the power conditioning modules and measuring the input power to the ion engine load bank as various control signals are generated by the avionics subsystem simulator.

6.1.1.7 Propulsion System Assembly and Test

A detailed receiving inspection and checkout of the various assemblies of the power subsystem is performed at the assembly site (JPL). Liquid metal loop integrity, continuity of all heater, control and instrumentation circuits, and design operation of the reactor control drums are confirmed. If the thrust subsystem has been stored for an extended time period, a checkout of its electrical circuits is performed.

The power and thrust subsystems are joined mechanically at the shield-thruster bay juncture. All of the power subsystem control and instrumentation cables are connected and the low voltage reactor cables connected to the bus bars of the thruster bay. All circuits joined are checked for continuity and shorts to the spacecraft structure.

Special facilities and equipment required for testing of the propulsion system include fixtures and cradles, plus a facility for the testing of the self-deployment and latching apparatus of the spacecraft (required only for the side thrust NEP Stage configuration). Also, an avionics subsystem simulator, capable of duplicating all the control signals for the power subsystem and the thrust subsystem, is needed to check the response of those subsystems to the expected range of command signals.

If necessary, the propulsion system is prepared for temporary storage.

6.1.1.8 Avionics Subsystem/Interplanetary Science Payload Fabrication, Assembly and Test

The avionics subsystem/interplanetary science payload is a collection of many specialized electronic assemblies and detectors, each of which is composed of numerous components. The individual experimental, control and communication assemblies are manufactured and performance tested at a number of different companies. These individual functional packages are then shipped to an assembly facility where they are mounted on the assemblies' structural frame or outer skin, which acts as a thermal radiation surface for cooling purposes.

The acceptance test for the avionics subsystem/interplanetary science payload requires a special simulator which duplicates the response and feedback of the propulsion system to command signals. Typical tests include the following:

1. Vibrate avionics subsystem/interplanetary science payload
2. Perform acceptance tests on monitoring and command assembly
    a. Simulate reactor startup to full power operation
    b. Check automatic control of propulsion system
    c. Check ground control of propulsion system
    d. Simulate reactor shutdown and restart
3. Test acceptance operation of communications assembly
    a. Check control and operation of antenna orientation mechanisms
    b. Check automatic and interrogation circuits of communications

4. Perform acceptance tests on navigation assembly

5. Perform operational acceptance checks of science payload

   a. Verify supply power of each instrument

   b. Confirm operation of output and communication circuits for each instrument

   c. Check control and operation of specific sensor orientation mechanism

6.1.1.9 Final NEP Stage Assembly and Test

A receiving inspection, testing each of the functional assemblies, is performed on the avionics subsystem/interplanetary science payload at the spacecraft-assembly facility. If the propulsion system has been stored for an extended time, a checkout of its major assemblies is made.

The avionics subsystem and science payload are attached to the propulsion system and all corresponding electrical cables are connected. A continuity check of each of the circuits is made. The spacecraft is installed in a special test support rig.

The acceptance test is performed as follows:

1. Vibrate spacecraft and check for mechanical damage.
2. Test spacecraft folding and unfolding operations (for applicable designs).
3. Check operation of translator mechanism and gimbals.
4. Perform leak tests on propellant subsystem and power subsystem.
5. Check individual operation of reactor control drums.
6. Check reactor instrumentation.
7. Check performance of propulsion subsystem control system.
8. Check mechanism controlling orientation of antennas and experiment sensors, if appropriate.
9. Check ground control of propulsion system.
10. Simulate reactor startup, shutdown, and restart operations.

11. Check automatic and interrogation circuits of communications.

12. Perform operational checks of scientific instrumentation.

With the completion of the acceptance tests, the propellant tanks are drained and safety locks on control drums are checked. The spacecraft is prepared and packaged for shipment to the launch facility.

6.1.2 PRELAUNCH OPERATIONS

Prelaunch includes all operations at the launch site up to lift-off. The major activities are:

1. NEP Stage inspection and systems checkout
2. Centaur inspection and systems checkout
3. Assembly of the Shuttle Payload Module
4. Shuttle servicing and checkout
5. Shuttle loading and mating
6. Readiness checks and Centaur fueling
7. Launch countdown

Figure 6-2 depicts the prelaunch operations.

6.1.2.1 NEP Stage and Centaur Checkout

A receiving inspection of the NEP Stage, performed at the launch site, includes operational checkouts of the power subsystem, the thrust subsystem, and the avionics subsystem. The checkouts require electrical power supplies which simulate reactor output, and electrical loads which duplicate ion engine power usage.

The Centaur stage is inspected and checkouts performed on its propulsion control, attitude control, and communications systems.

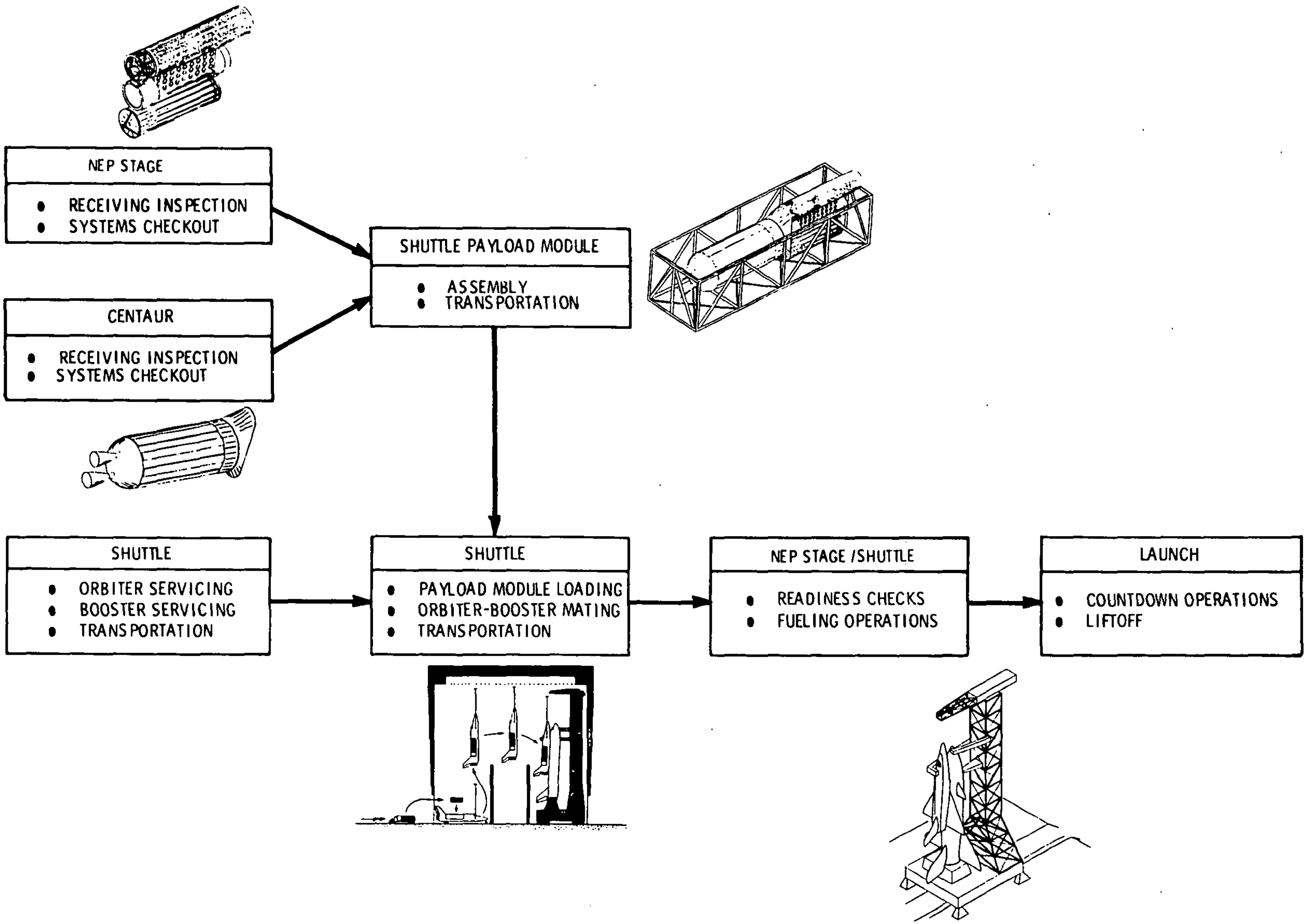


Figure 6-2. Phase 2: Prelaunch Operations

6.1.2.2 Shuttle Payload Module Assembly

The payload module consists of the Centaur and NEP Stage plus payload joined together and installed in a Transfer Module. The NEP Stage is first clamped in the transfer module and then its propellant tanks are filled with mercury. After pressurization and leak checks, the propellant charging ports are seal welded. The Centaur vehicle then is joined structurally to the NEP Stage and clamped in place in the transfer module.

An auxiliary power supply assembly, if included in the design, is installed on the transfer module and connected electrically to the Centaur/NEP Stage. A check is made for proper operation of the transfer module release clamps and mechanisms. A special transporter then transfers the payload module to the Shuttle assembly building with necessary environmental control protection and monitoring of the NEP Stage conditions.

6.1.2.3 Shuttle Checkout and Loading

Prior to loading of the Shuttle, servicing and systems checkout of the booster and Shuttle orbiter are performed in the standard manner and location. Then the orbiter and booster are moved to the Shuttle assembly area where the Payload Module is loaded into the cargo bay of the orbiter and umbilical cables for the monitoring of the Centaur/NEP Stage are attached. The booster is checked out for proper operation of orbiter attachment points, then prepared for the mating procedure. The orbiter is attached to the booster and the mated Shuttle configuration is transferred to the launch pad by standard Shuttle procedures and equipment. In the meantime, launch pad facilities, including specialized equipment for the NEP Stage are activated.

6.1.2.4 Readiness Checks and Launch

The final launch pad operations are readiness checks, fueling operations and countdown. The individual activities will include the following:

1. Perform spacecraft readiness checks.

    a. Check NEP Stage subsystems.

b. Check Centaur systems.

c. Perform simulated launch tests.

2. Perform fueling operations.

a. Deactivate all spacecraft electrical systems.

b. Fuel Centaur Stage.

c. Remove mechanical safety locks from reactor control drum mechanisms.

d. Seal space shuttle cargo bay and flood with inert gas.

e. Fuel orbiter and booster.

f. Heat up NEP Stage liquid metal loops using auxiliary power supply in cargo bay.

3. Countdown and range safety approval

6.1.3 LAUNCH AND EARTH ORBIT

The Launch and Earth Orbit Phase encompasses the activities from time of lift-off until earth escape operations are ready to begin. The major activities are:

1. Shuttle ascent
2. Spacecraft release operations
3. Spacecraft release in earth orbit

The major activities during this mission phase are illustrated in Figure 6-3.

6.1.3.1 Ascent

After lift-off, the Shuttle ascent operations follow standard Shuttle procedures. Booster-orbiter separation is followed by a coast period to the apogee of the initial orbit. At that point, the orbiter engines are ignited for the first burn of a Hohmann transfer to the final orbit. The burn is followed by a coast period to the apogee of the transfer orbit where a second orbiter burn circularizes the final orbit at a 500 km altitude. During the ascent phase, NEP Stage system conditions are continuously monitored by the orbiter crew and ground station personnel.

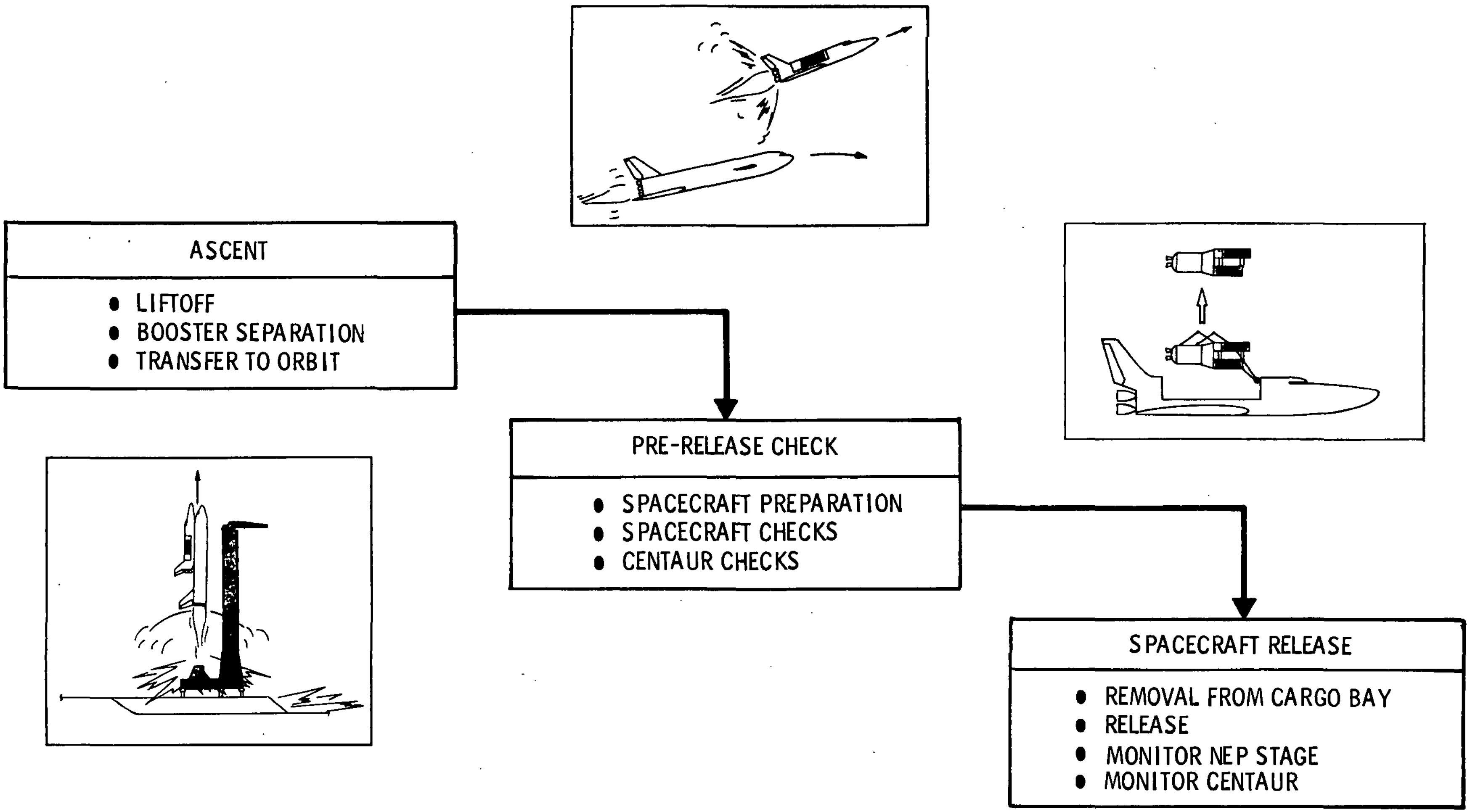


Figure 6-3. Phase 3: Launch and Earth Orbit

6.1.3.2 Pre-Release Checkout

The initial activities in the parking orbit prepare the NEP Stage and Centaur for release and confirm operation of each of the NEP Stage Subsystems. The sequence of events is:

1. Heat stage subsystems to desired temperatures using Shuttle auxiliary power supply in cargo bay.
2. Activate stage subsystems using Space Shuttle auxiliary electrical power.
3. Check operations of NEP stage subsystems.
   a. Reactor
   b. Propellant feed system
   c. Ion Engine translator and gimballing mechanisms
   d. Communication equipment
   e. Science payload equipment
   f. Spacecraft attitude control equipment
   g. All instrumentation and monitoring circuits
   h. NEP Stage auxiliary power supply
4. Switch stage systems to stage auxiliary power supply
5. Activate and checkout Centaur.

The failure of any assembly, subsystem, etc., vital to the successful completion of the mission, requires the return of the NEP Stage to the Shuttle orbiter.

6.1.3.3 Spacecraft Release

The removal and release of the Centaur/NEP stage from the Shuttle orbiter is accomplished in the following steps:

1. Open cargo bay doors of the orbiter and remotely grapple Centaur/NEP stage with manipulator arms.

2. Remove payload from cargo bay to extent position of manipulators.

3. Orient NEP stage/Centaur to initial conditions required for injection to earth escape (attitude, etc.) and release.

4. Monitor NEP stage and Centaur functions.

5. Secure manipulator arms and back orbiter away from Centaur/NEP stage.

6.1.4 EARTH ESCAPE AND NEAR EARTH OPERATIONS

This phase of the mission covers the operations from the initiation of earth escape chemical propulsion to the beginning of full power electric propulsion. The major activities are:

1. Centaur stage burn and separation from the NEP stage

2. NEP stage startup operations

3. Initiation of electric propulsion

Figure 6-4 shows the key mission operations that occur during the Earth Escape and Near Earth Operations Phase.

6.1.4.1 Centaur Operations

During the earth escape operations, the Centaur/NEP stage is under the direction of the Centaur Control systems. Precise orbit parameters are determined by ground control facilities, and corrections for stage attitude and orientation are communicated to the Centaur. Burn initiation and duration are automatically sequenced by the Centaur systems, except if modified by ground control. After Centaur shutdown, the attitude control system reorients the NEP Stage as necessary.

6.1.4.2 NEP Stage Operations

Three critical operations are performed in this subphase: NEP stage deployment, NEP stage/ Centaur separation, reactor and power subsystem startup, and ion engine startup. The first, NEP stage deployment, occurs following Centaur shutdown and stabilization. Successful deployment and latching is followed by reorientation of the stage for acceptable communications

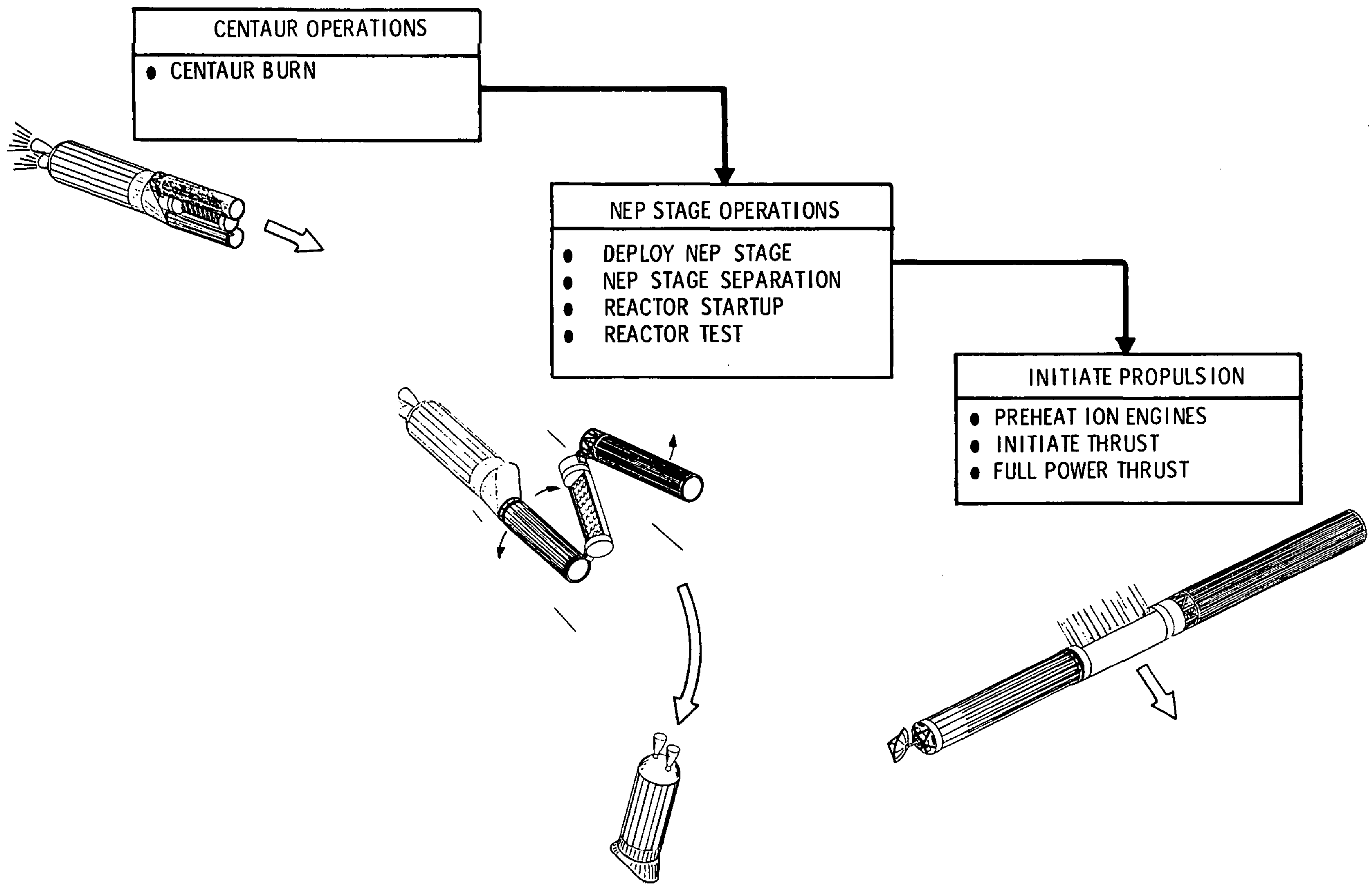


Figure 6-4. Phase 4: Earth Escape and Near Earth Operations

with ground control,and the attitude control functions are transferred to the NEP stage control system. Then the Centaur and adapter section separates from the NEP Stage. The Centaur remains attached to the NEP Stage throughout the deployment operation to provide the necessary attitude control and stabilization.

The NEP Stage deployment operations pertain only to the side thrust NEP Stage configuration. The end thrust stage configuration requires no deployment following the Centaur burn. After the high energy burn, the Centaur is released from the end thrust NEP Stage by firing pyrotechnic separation devices located on the internal support mating ring. This separation scheme could include a spring-activated separation mechanism. The Centaur is guided out of the NEP Stage by a four-rail guide system to ensure minimum angularity differences during separation.

The reactor and power subsystem operation is started with the following procedure:

1. Initiate reactor startup to zero power.
   a. Establish desired coolant flow rates and pressure level.
   b. Adjust control drums incrementally and confirm neutron multiplication factors.
2. Bring powerplant to self-sustaining power level.
   a. Automatically sequence drum rotation to achieve idle power conditions in reactor.
   b. Adjust cesium reservoir temperature conditions as necessary.
   c. Switch hotel loads from auxiliary power to reactor power.
   d. Switch all avionic subsystem functions except instrumentation and control to reactor power.
   e. Recharge auxiliary power system, if appropriate, from reactor.
   f. Adjust reactor power for steady-state operation at idle condition and monitor power subsystem conditions.

When satisfactory conditions exist in the power subsystem, a high power test is performed

by increasing and stabilizing reactor power at a power level producing design temperatures on the diode emitters. Open circuit voltage across TFE pairs is checked, as well as temperatures, flow conditions, power and temperature distributions, etc., in the reactor.

6.1.4.3 Initiate Electric Propulsion

With satisfactory operation of the reactor and power subsystem established, the ion engine array is activated. The first step is to preheat the ion engines and vaporizer sections with electrical power taken from the reactor output, adjusting the reactor power level as necessary. When suitable temperatures are reached in the ion engines, the propellant flow to the engines is started and the NEP Stage is adjusted to the orientation required for electric propulsion.

Thrust is produced from the ion engines by switching on screen current to the ion engines, two at a time, in sufficient quantity to produce ~ 25 percent thrust. The rate of ion engine startup is limited to the maximum power ramp allowed in the reactor. Once all the engines have been started, four engines are turned off for standby status. The final operation is the gradual increase in screen current on all thrusters, simultaneously, to full power conditions. Again, the rate of increase will depend on allowable reactor power ramp rates.

6.1.5 HELIOCENTRIC FLIGHT

The mission operations in this phase cover the longest time span of the mission, namely, the heliocentric flight period from immediately after earth escape until approach of the target planet/comet is achieved. The major activities (see Figure 6-5) include the following:

1. Stabilization of system conditions in the NEP Stage
2. A continual schedule of navigation, experimentation, system checks and communications
3. A mid-course coast period
4. A second period of electric propulsion with ion engine restart and a continual schedule of navigation, etc.

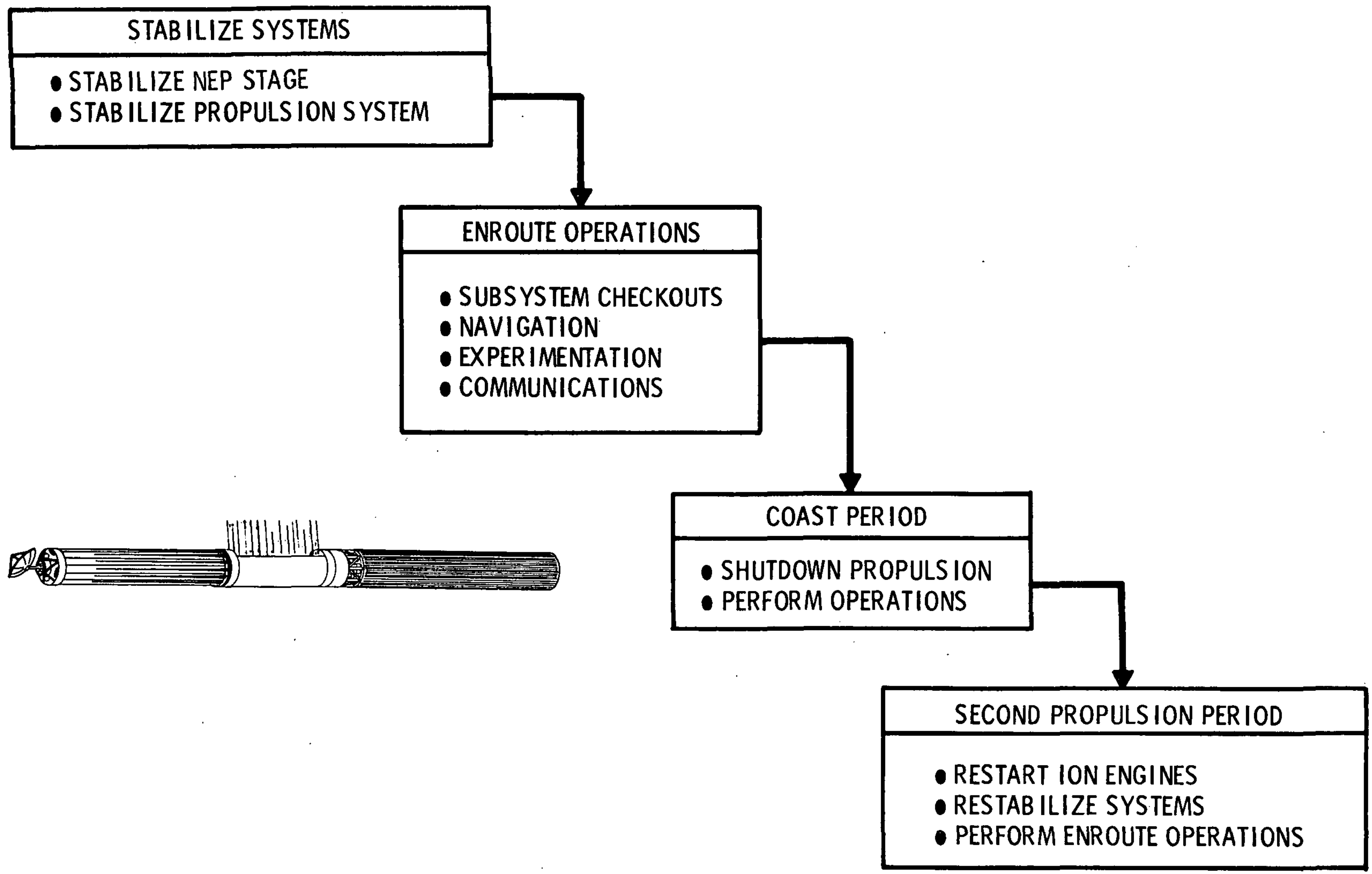


Figure 6-5. Phase 5: Heliocentric Flight

6.1.5.1 Stabilize Systems and Establish Enroute Operations

Immediately following the establishment of electric propulsion, altitude control is switched to the ion engine array. Navigation data is received from ground control and NEP Stage attitude/orientation is corrected as necessary to adjust acceleration vector of the spacecraft. Power subsystem conditions are monitored automatically by the on-board computer, and adjustments are made on reactor power setting, coolant flow rate, cesium reservoir temperatures, etc.

A continual schedule of operations is established as follows:

1. Periodic checkout of NEP Stage subsystem operations
   a. Every hour check communication channels
   b. Every four hours check the following:
      (1) Operation of spacecraft attitude measurement and control system
      (2) Operation of navigation system
      (3) Data processing operations of the computer
   c. Every twenty-four hours check the following:
      (1) Operations of the nuclear power system
      (2) Diagnostics for computer operation
2. Periodic navigation operations
   a. Every hour check NEP Stage attitude and correct if necessary
   b. Every twenty-four hours determine navigation position of the NEP Stage and reorient spacecraft in direction of required acceleration vector
   c. Every twenty-four hours realign the inertial navigation unit
   d. Rebalance stage by transference of mercury propellant between tanks as necessary (required only for side thrust configuration)
3. Activate appropriate scientific experiments

a. Orient sensors of experiments, if necessary

b. Activate data processing function of computer

c. Activate automatic telemetry of experimental data

4. Initiate and maintain communications schedule

   a. Continuous transmission of raw and/or processed data from experiments

   b. Real time transmission of intermittent experiment data

   c. Transmission of results of periodic checkouts

   d. Transmission of results of periodic navigation operations

   e. Continuous monitoring of ground-based signals

6.1.5.2 Coast Period

The major activities of the mid-course coast period are the shutdown of electric propulsive thrust and the continuation of navigation, experimentation etc., operations. The ion engine deactivation is performed in the following sequence:

1. Switch NEP stage attitude controls to secondary mode of operation.

2. Shut down electric propulsion.

   a. Turn off mercury propellant flow and switch off electrical power in ion engines in mirror image pairs.

   b. Concurrent with above, reduce reactor power level to maintain constant power subsystem temperature.

   c. Stabilize power subsystem at idle conditions.

The NEP Stage attitude is periodically checked and corrected, if necessary, to maintain desired orientation for experimentation. Communications will be maintained as in the electric propulsion period.

6.1.5.3 Reestablish Electric Propulsion

Restart of the ion engines is the first activity of the second propulsion period and it follows the same procedure as the initial engine start. The NEP Stage is reoriented, if necessary, the ion engines are preheated, propulsion is initiated in opposing pairs of ion engines to the ~ 25 percent level until all the engines are started, and full thrust is attained by increasing the screen current to full design levels in all the engines simultaneously. The stage subsystems are stabilized at design conditions and a periodic navigation schedule with vector correction is established as in the first propulsion period. The other periodic operations continue as before.

6.1.6 PLANET/COMET ARRIVAL

This phase covers the final operations of the flight at the target planet or comet. For a planetary mission, Phase 6 begins when the NEP Stage enters the gravitational influence of the target planet. The major activities include:

1. Special navigation and experimental operations as the planet is approached.
2. Special experimentation as the NEP Stage spirals in through the planet atmosphere.
3. Propulsion system shutdown and planet survey experimentation at designated orbiting altitude.

For the cometary mission, Phase 6 starts when the special unit on board the NEP Stage detects the oncoming comet. The major activities in this phase of a comet mission include:

1. Course corrections and special experimentation as the NEP Stage approaches the comet.
2. Propulsion system shutdown and comet survey experimentation.

The key mission operations involved in the Planet/Comet Arrival Phase are illustrated in Figure 6-6.

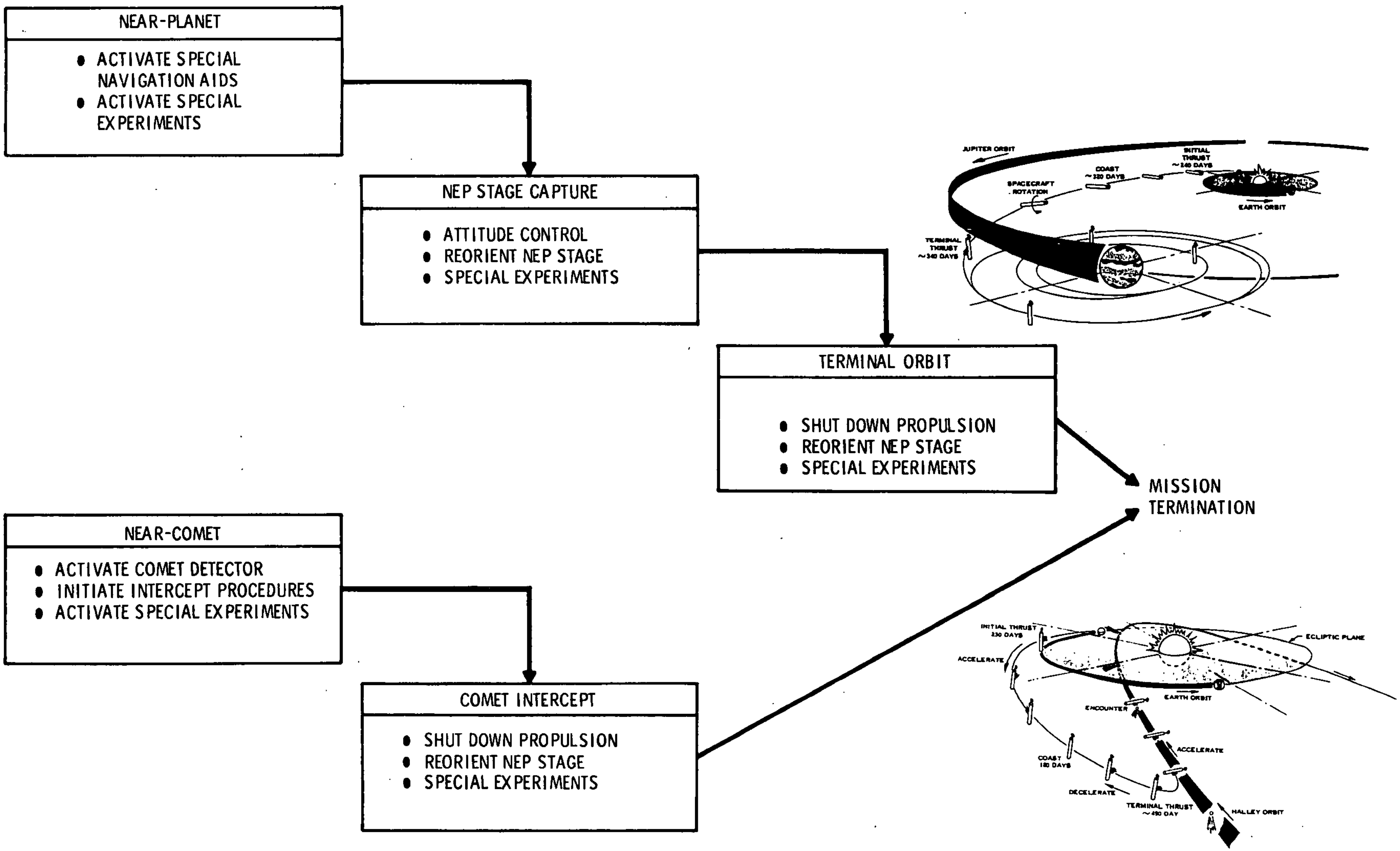


Figure 6-6. Phase 6: Planet/Comet Arrival

6.1.6.1 Planet Arrival

As the NEP Stage nears the target planet, the radar or planet detector unit is activated and its data utilized to correct the intercept trajectory of the stage. Special experimentation, if appropriate, is activated.

As the NEP Stage enters planetary orbit, attitude and thrust vector control is switched to a planet-oriented control system. The stage is reoriented and experimentation of planetary atmosphere is activated. Communication scheduling is revised, if necessary to account for line-of-sight blockage by the planet.

When the NEP Stage reaches the final orbit altitude, the attitude control is switched to the alternate system. The electric propulsion engines are shut down and reactor power output stabilized at the terminal operating level. The stage is reoriented and planet surveying instrumentation activated. Experimentation and communications continue until experimentation objectives are achieved.

6.1.6.2 Comet Arrival

When the comet detector unit "sees" the approaching comet, the NEP Stage is reoriented, and navigational control is switched to the detector. Special experimentation is activated, if appropriate.

Upon comet interception, attitude control is switched to the alternate mode, and the NEP Stage is reoriented to expedite comet observations and corresponding communications. The propulsion engines are shut down and the power subsystem stabilized at the required generating level. Experimentation and communications proceed on a continuous basis until experimental objectives have been satisfied.

6.2 GEOCENTRIC MISSIONS

This section summarizes and illustrates the mission profile and operations for the NEP Stage in geocentric orbit mission application. The scope of this section covers mission operations starting with delivery of the NEP Stage to the launch site, through end-of mission disposal of

the NEP Stage in heliocentric orbit. The Fabrication and Test Phase is assumed to be the same as that already described in Section 6.1.1.

6.2.1 BASELINE MISSION OPERATION

The baseline mission profile the NEP geocentric orbit operations is presented in Figure 6-7. The baseline mission for the 120 kWe NEP Stage is composed of the following major elements:

1. The NEP Stage and the attached Propellant Logistics Depot (PLD) is placed into a 28.5 degree low earth orbit (~435 km) by the Space Shuttle (Shuttle Launch 1).

2. The NEP Stage transports the PLD to a 14,800 km by 35,800 km parking orbit (15 degrees inclination) using electric propulsion (trip time required is approximately 145 days). The PLD is detached from the NEP Stage and remains in the parking orbit to provide future logistic support.

3. The Chemical Tug, with a synchronous orbit payload attached, is delivered to a 28.5 degree low earth orbit (~435 km) by the Space Shuttle (Shuttle Launch 2).

4. The Chemical Tug transfers the synchronous orbit payload to the parking orbit. Rendevous with and transfer of the payload to the NEP Stage is accomplished.

5. The Chemical Tug returns to low earth orbit, docks with the Shuttle, and is returned to earth for re-use.

6. The NEP Stage transports the payload to a 35,800 km synchronous equatorial orbit and deploys it. (The NEP Stage has the capability, ~ 8600 kg in this mission profile, to deliver multiple payloads to synchronous orbit).

7. The NEP Stage returns to the lower parking orbit, docks with the PLD, and refuels.

8. A second synchronous orbit payload is placed in the parking orbit by the Chemical Tug (Shuttle Launch 3).

9. The synchronous orbit payload is docked to the NEP Stage.

10. The Chemical Tug returns to earth.

11. The NEP Stage delivers the payload to synchronous orbit.

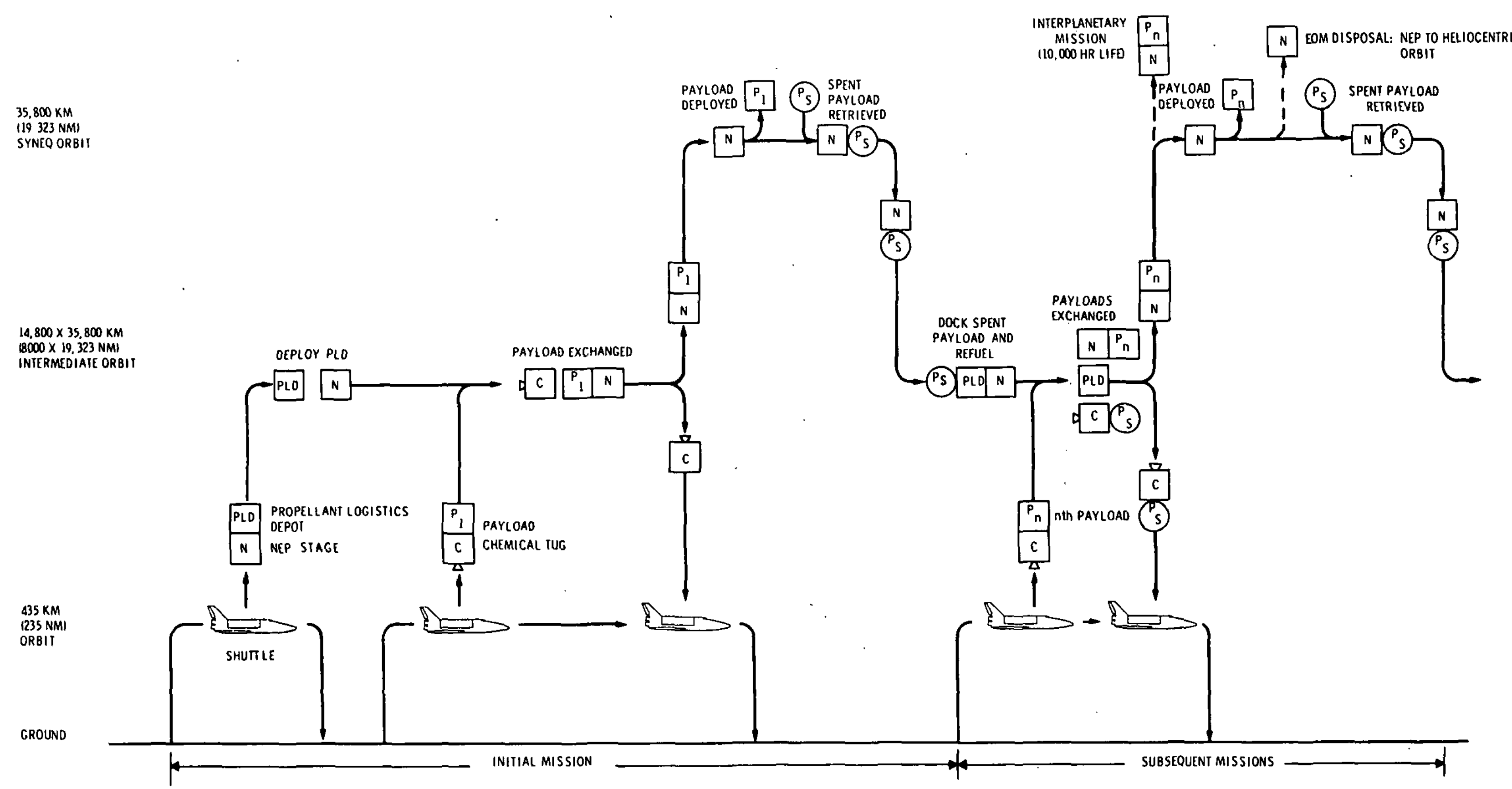


Figure 6-7. Baseline Geocentric Orbit Mission Profile

12. Upon deployment of the synchronous orbit payload, the NEP Stage may rendezvous and dock with a "spent" synchronous payload and return it to the lower parking orbit. (It may also return empty; return trip time increment for the empty return mode is approximately 15 days).

13. The NEP Stage docks the spent payload to one end of the PLD, thereby maintaining positive handling control of the spent payload.

14. The NEP Stage docks with the other end of the PLD. When refueling is completed, the NEP Stage undocks and remains in the parking orbit, in close proximity to the PLD.

15. Another synchronous orbit payload is placed in the parking orbit by the Chemical Tug (Shuttle Launch N).

16. The Chemical Tug docks the new payload to the NEP Stage.

17. The Chemical Tug docks with the spent payload attached to the PLD. The Chemical Tug/payload system undocks from the PLD.

18. The Chemical Tug and the spent payload return to low earth orbit. Shuttle rendezvous and earth return is accomplished.

    The mission profile is repeated until the NEP Stage almost completes its 20,000 full power hour design life (~ 10 missions, depending on the mass of the synchronous orbit payloads).

19. At the end of this period and after transportation of the n^{th} payload to synchronous orbit, the NEP Stage is assumed to dispose of itself via spiral to earth escape into heliocentric orbit.

The following subsections further detail the NEP Stage geocentric orbit mission operations. Key events discussed are:

1. Arrival of the NEP Stage at the launch site
2. Prelaunch operations
3. Launch and deployment of the NEP Stage
4. NEP Stage - PLD transfer to the parking orbit
5. Propellant Logistics Depot deployment

6. Chemical Tug/synchronous orbit payload launch

7. Rendezvous between Chemical Tug and NEP Stage payload transfer

8. Chemical Tug return to earth

9. Placement of payload in synchronous orbit

10. NEP Stage retrieval of spent synchronous orbit payload (and subsequent refueling operations)

11. NEP Stage places new (second) synchronous orbit payload in orbit

12. NEP Stage end-of-mission disposal

6.2.1.1 Arrival of NEP Stage at Launch Site

This mission phase begins with the arrival of the NEP Stage at the launch site. During this phase, a series of functional checks will be performed to ensure that the NEP Stage has not been damaged enroute to the launch site. In addition, a Propellant Logistics Depot (PLD) will be mated to the stage at the launch site, before installation in the Shuttle. The PLD is placed in the reference parking orbit by the NEP Stage and contains all the mercury propellant and other consumables required over the operational life of the NEP Stage. Special facilities required for this mission phase include a nuclear test facility, alkali metal handling capability, mercury propellant handling facilities and nuclear radiation instrumentation.

It is assumed that the NEP Stage arrives at the launch site completely intact, enclosed in a special shipping container. The containment vessel must be designed to:

1. Minimize shock and vibration on the spacecraft

2. Prevent flooding of the reactor system in the event of water immersion

3. Provide acceptable temperature and humidity control

4. Facilitate handling of the NEP Stage

Subsequent to visual inspection of the NEP Stage, a series of tests will be initiated. These activities will include a check of the reactor control mechanisms, fluid flow systems, propulsion system checkout, and operation of the avionics subsystem. Testing of the reactor control system will consist of subcritical operations during which individual control drums will be rotated. Fluid flow tests will check out the operation of EM pumps and the condition of flow channels. Checkout of the thrust subsystem will consist of electrical continuity tests and thruster gimballing mechanisms. An assessment of the operation of the avionics subsystem which contains the attitude control, communications, docking and other subsystems will also be performed. All of these tests will be conducted in a nuclear storage and test building. The shipping container will be designed such that it will be unnecessary to remove the NEP Stage from the container during the performance of these tests.

6.2.1.2 Prelaunch Operations

The prelaunch operations will consist of attaching the PLD to the NEP Stage and placing the total system in the cargo bay of the Shuttle while inside the VAB. From this point the Shuttle is transported to the launch pad for the initial launch involving the NEP Stage. These operations will require a transporter to take the NEP Stage from the nuclear test building to the orbiter, and a handling system to install the NEP Stage and PLD in the Shuttle cargo bay.

Following mating of the Shuttle orbiter to the booster, the mated configuration is taken to the launch pad where final continuity and integration checks are made before launch.

6.2.1.3 Launch and Deployment of NEP Stage

Following the launch of the Space Shuttle, the orbiter transports the NEP Stage to a 435 km low earth orbit where it deploys the NEP Stage. Therefore, the cargo bay of the orbiter must be equipped with a deployment mechanism for the NEP Stage and the PLD payload. Pre-startup checkout of the NEP Stage will occur before its deployment from the Shuttle.

6.2.1.4 NEP Stage/PLD Transfer to Intermediate Parking Orbit

Following deployment of the NEP Stage, the propulsion system must be started. The orbiter will have the capability to monitor the startup procedure (and possibly aid in the correction of any operational difficulties). In the event that startup of the NEP Stage cannot be effected, the orbiter will have the capability to reacquire the NEP Stage.

Startup of the NEP Stage propulsion system will be initiated by startup of the reactor. During this period, electrical power to the reactor control drums and EM pumps will be supplied by the startup auxiliary power supply nickel-cadmium batteries. As the reactor approaches operating power, the mercury propellant is heated and electrical energy is supplied to the thrusters.

The NEP Stage with its mercury propellant payload begins to spiral out from the earth, finally attaining a 14,800 km by 35,800 km elliptical orbit. At this time, the mercury propellant depot is deployed in this intermediate orbit for subsequent logistic support. The PLD must be equipped with a signalling device to facilitate docking and tracking. In addition, an attitude control system must be provided aboard the depot to facilitate future docking maneuvers.

6.2.1.5 Propellant Logistics Depot (PLD) Deployment

The NEP Stage will take approximately 145 days to achieve the reference elliptical parking orbit from low earth orbit, while transporting the approximately 7,400 kg PLD payload. Following attainment of the intermediate parking orbit, the NEP Stage undocks from the PLD. The NEP Stage must have the capability to redock and undock from the PLD for future refueling operations.

The NEP Stage will remain in this intermediate parking orbit near the PLD until a synchronous orbit payload is brought up to the parking orbit by the Chemical Tug. The NEP Stage is partially shut down during this time.

6.2.1.6 Synchronous Orbit Payload Launch

The mission objective is to place an operational payload into synchronous orbit. This procedure is initiated with the arrival of the synchronous payload at the launch site where it is mated to the Chemical Tug. The Chemical Tug, with the payload attached, is placed in the Shuttle (Launch 2), using a transfer module similar to the one provided for the NEP Stage. Other facilities which are required for this mission phase include deployment facilities for the Chemical Tug/synchronous payload aboard the Shuttle.

The Shuttle takes the Chemical Tug and attached payload to the 435 km low earth orbit where it is deployed. Startup of the Chemical Tug is monitored by the orbiter. The Chemical Tug propels itself into the reference intermediate parking orbit where rendezvous with the NEP Stage is accomplished. Approximately five to six hours will be required for the Chemical Tug to reach the parking orbit.

6.2.1.7 Rendezvous Between Chemical Tug and NEP Stage - Payload Transfer

During this operation, the NEP Stage acquires the first synchronous payload from the Chemical Tug. For this payload transfer, ground support in the area of tracking and guidance will be required to orient the vehicle properly for the docking and detachment procedures. This docking procedure is dictated by the requirement that the synchronous orbit payload remain under positive handling control at all times.

6.2.1.8 Chemical Tug Return to Earth

Following detachment from the payload, the Chemical Tug returns to the 435 km low earth orbit. Once in low earth orbit, the Chemical Tug proceeds to rendezvous with the Shuttle. The Shuttle returns the Chemical Tug to the ground where it undergoes refurbishment.

6.2.1.9 Placement of Payload in Synchronous Orbit

After acquiring the synchronous orbit payload, the NEP Stage is started up, and begins the process of orbit circularization into an equatorial synchronous orbit. Once this orbit is achieved, the NEP Stage detaches from the synchronous payload.

Before returning to the lower parking orbit, the NEP Stage may rendezvous with a spent payload, and return it to the intermediate parking orbit for subsequent return to earth. This mission event is assumed for this operational sequence definition. The NEP Stage could return empty to the lower parking orbit.

6.2.1.10 NEP Stage Retrieval of Spent Payload

After placing a new payload into synchronous orbit, the NEP Stage will normally be used to retrieve a spent payload which will be returned to earth by transfer of the Chemical Tug and then to the Shuttle. The facilities which will be required for these series of maneuvers are a docking system aboard the mercury propellant logistics depot, and ground support equipment to effect several docking procedures which are subsequently described.

The NEP Stage will be in synchronous orbit following deployment of the new payload. Therefore, link-up with the spent synchronous payload will require minor orbit change. Following retrieval of the spent synchronous payload, the NEP Stage will spiral down into the intermediate parking orbit. The NEP Stage, with the spent synchronous payload attached, will rendezvous with the propellant logistics depot. In order to provide positive handling and transfer procedures, the transfer of the spent synchronous payload to the Chemical Tug will occur in the following manner:

1. The NEP Stage will temporarily dock the spent synchronous payload to the mercury propellant depot (PLD).

2. The NEP Stage will undock from the spent synchronous payload, and dock directly to the opposite side of the PLD.

3. While waiting for the Chemical Tug, the NEP Stage will conduct refueling operations and upon completion, detach from the PLD.

4. The Chemical Tug, carrying a new synchronous payload, completes rendezvous with the NEP Stage which acquires the new synchronous payload.

5. The Chemical Tug, devoid of the new synchronous payload, docks to the spent payload which is still attached to the PLD.

6. The Chemical Tug/spent synchronous payload detaches from the PLD and returns to the 435 km orbit for Shuttle rendezvous.

6.2.1.11 NEP Stage Deploys New Synchronous Orbit Payload

The NEP Stage with the new synchronous payload attached, begins circularization of the parking orbit and plane change until equatorial synchronous orbit is reached. Deployment of the new payload will proceed as previously specified in Section 6.2.1.9.

The procedure of retrieving a spent payload, subsequent to placing a new payload in orbit, and returning to the PLD will be repeated until the useful life of the NEP Stage is expended.

6.2.1.12 NEP Stage End-of-Mission Disposal

The NEP Stage will insert approximately 10 payloads into synchronous orbit; at this time, the 20,000 hour useful life of the NEP Stage will be expended and it must be disposed of. Several options for safe disposal of the NEP Stage are available. The recommended approach is to have the NEP Stage insert itself into a heliocentric orbit. This orbit transfer can be accomplished with the existing NEP Stage propulsion capability.

6.2.2 ALTERNATE GEOCENTRIC MISSION PROFILES

In addition to the baseline NEP Stage geosynchronous orbit mission, several alternate missions have been identified. Two mission modes for the fast delivery (~6 hours) of payloads to synchronous equatorial orbit are depicted in Figure 6-9.

The first mission mode involves the Chemical Tug transporting a synchronous orbit payload and NEP Stage to geosynchronous orbit. The payload is deployed and the NEP Stage is used to return the spent Chemical Tug to the intermediate parking orbit for return to the Shuttle by the next Chemical Tug sortie.

Another fast delivery mission mode again involves the payload being transported to geosynchronous orbit by the Chemical Tug. After the payload has been deployed, an NEP Stage (which has been waiting in geosynchronous orbit since deploying a payload of its own) rendezvouses with the Chemical Tug, docks, and returns the spent Chemical Tug to the intermediate orbit for return to the Shuttle by the next Chemical Tug sortie.

In both of these mission modes, the option exists for the NEP Stage to return the spent Chemical Tug to low earth orbit directly rather than to the intermediate parking orbit.

The all-NEP mission (see Figure 6-9) represents another NEP Stage geosynchronous mission alternative. In this mission mode, the NEP Stage with payload spirals out to geosynchronous orbit and back with no chemical assist.

Van Allen radiation protection will be required for the power conditioning electronics and certain Net Stage electronics. Depending on the spiral out time through the radiation belts, the synchronous orbit payload may also require electron and proton radiation protection.

6.3 DUAL MISSION MODE

The multi-mission NEP Stage has the capability to perform in a dual mission mode. In this type of operation, the NEP Stage performs approximately three to five geocentric orbit missions, then performs a 10,000 hour interplanetary mission.

In performing the geocentric orbit missions, the mission operations are the same as those just discussed in Section 6.2. Upon transporting the next to last operational payload to synchronous orbit and returning to the 14,800 by 35,800 km intermediate parking orbit, the NEP Stage refuels from the PLD. After taking on enough propellant to transport a final payload to synchronous orbit and also to perform a 10,000 hour interplanetary mission, the NEP Stage rendezvouses with the Chemical Tug. On this final sortie, the Chemical Tug brings up an "integrated payload" to transfer over to the NEP Stage. This integrated payload (see Figure 6-10) consists of an operational synchronous orbit payload, an adapter truss assembly, and an interplanetary science payload with a parabolic communication

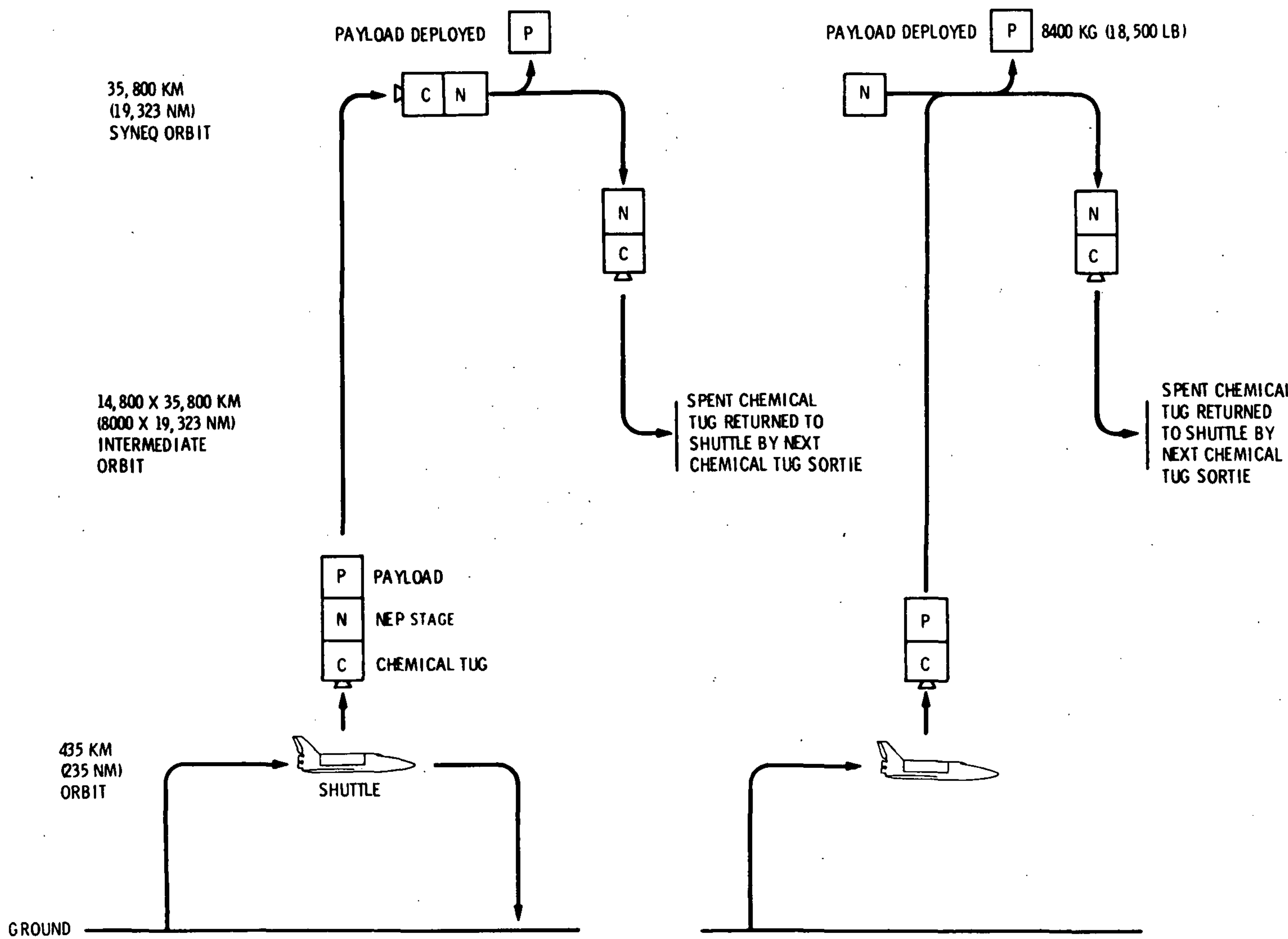


Figure 6-8. Alternate NEP Stage Geosynchronous Orbit Mission Modes (Fast Delivery ~ 6 Hours)

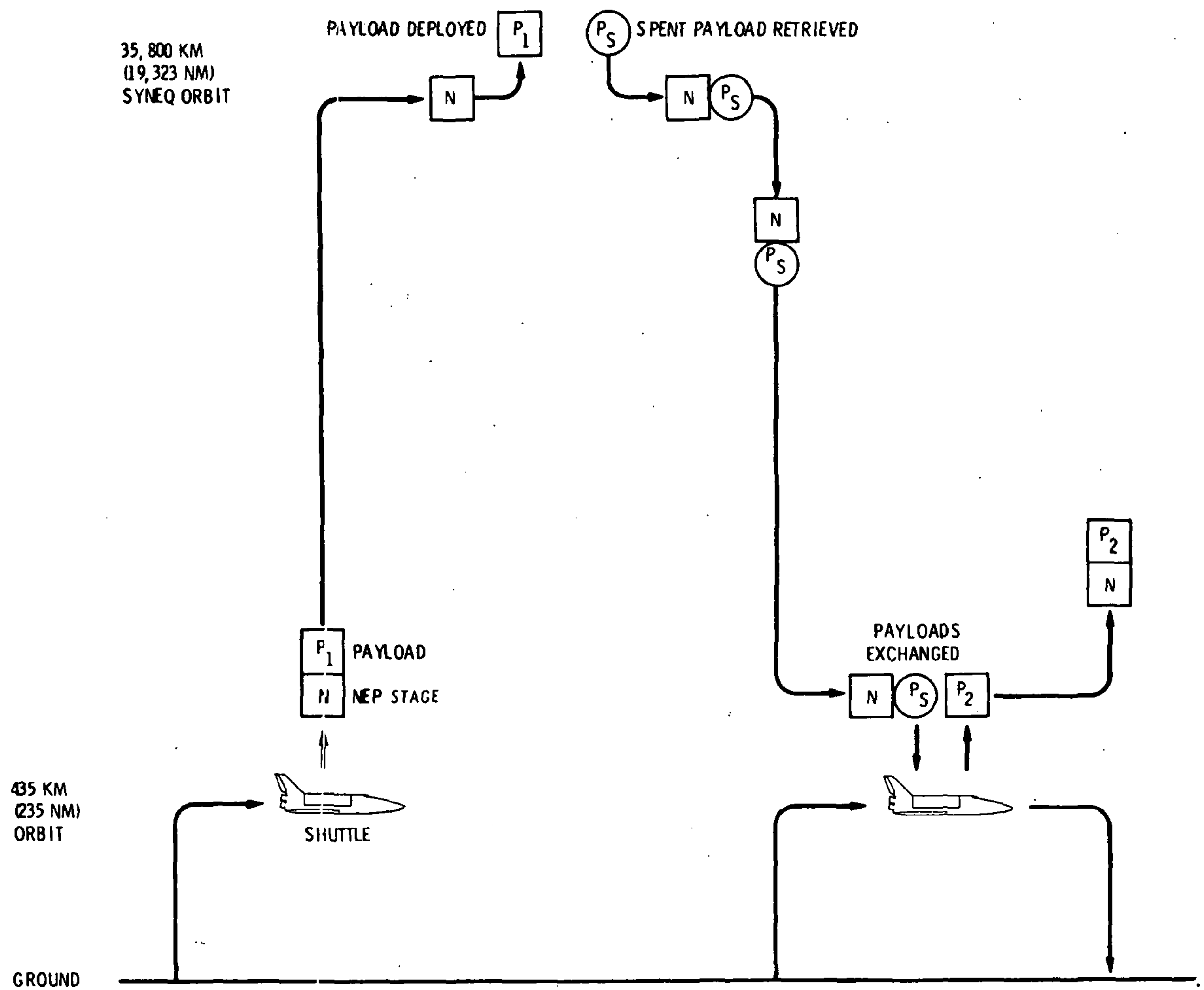


Figure 6-9. Alternate NEP Stage Geosynchronous Orbit Mission Modes (all NEP)

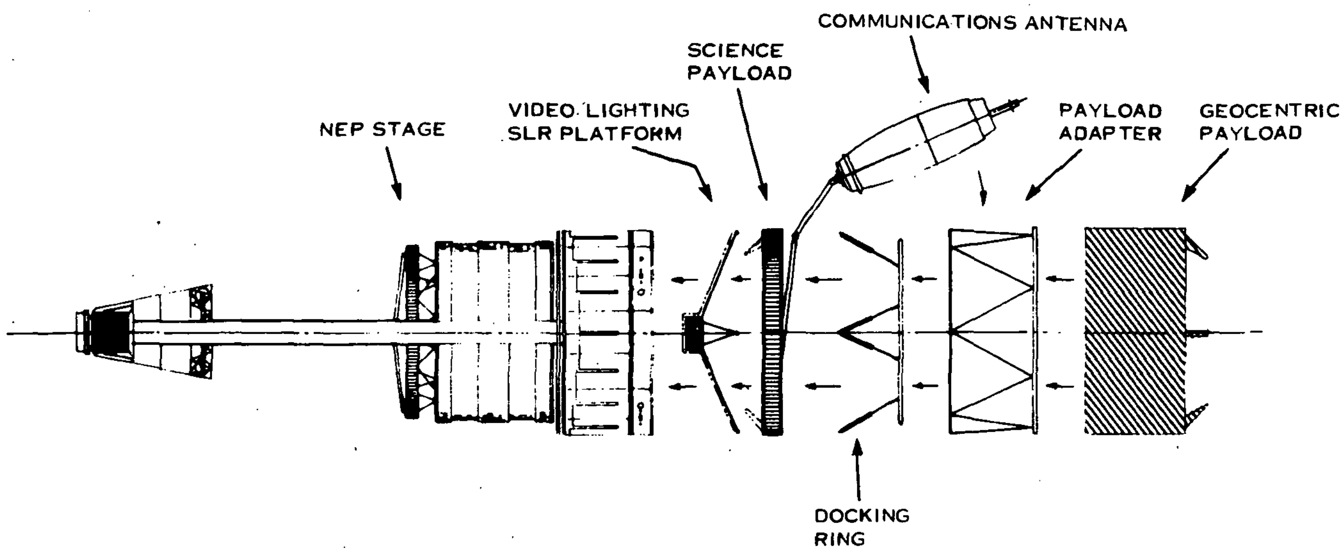


Figure 6-10. Integrated Payload for Dual NEP Mission Applications

antenna (in a stowed configuration). Upon receipt of this "payload", the NEP Stage spirals out to synchronous altitude and deploys the operational payload. The operational synchronous orbit payload is separated from the science payload by explosive bolts that are located on the adapter truss between the two payloads. Once the synchronous orbit payload is released, the parabolic communication antenna is deployed and the NEP Stage positions itself for a low thrust earth escape via electric propulsion. The remainder of this 10,000 hour interplanetary mission is the same as that discussed in Section 6.1.

SECTION 7

GSE AND OPERATIONAL EQUIPMENT

The mission profiles and operations presented in Section 6 identified various key Ground Support Equipment (GSE) and Operational Equipment necessary to support the NEP Stage operations. Those facilities and equipment that remain on the ground are included in GSE; whereas, Operational Equipment is limited to flight hardware.

7.1 GROUND SUPPORT EQUIPMENT

The key GSE items are:

1. Nuclear Storage and Checkout Facility

This facility provides for remote controlled environment storage and non-nuclear acceptance testing of NEP Stage system delivered to the launch site. The majority of the nuclear hardware prelaunch activities should be accomplished in an isolated facility capable of supporting testing and storage operations. Existing facilities at KSC such as the Pyrotechnic Installation Building located in the Industrial Area may meet future requirements of a single nuclear reactor payload. However, this facility is inadequate for processing and storage of several reactor systems.

A new facility, referred to as the Nuclear Storage and Checkout Facility is required at KSC for a program involving several large nuclear systems. The NS&C Facility should be capable of supporting a minimum of three nuclear systems (and several isotope heat sources) in various stages of assembly, test and storage. (Reactor and isotope storage must be separated from the assembly and test bays by suitable radiation shielded, blastproof and fireproof walls.)

The requirements and hazardous characteristics of reactor power modules differ significantly from those of an isotope heat source. A low nuclear and liquid metal hazard potential and low radiation exposures to personnel can be achieved by providing separate assembly areas for isotopes and reactor power modules where simultaneous operations can be performed.

A typical layout of such a facility and preliminary requirements are shown in Figure 7-1. A railroad spur is shown adjacent to the building to provide for transportation.

The area requirements of the NS&C Facility could be substantially changed, depending on multiple program usage.

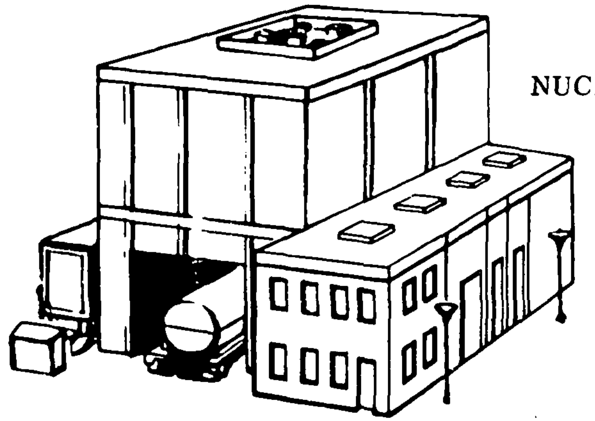
Location of the NS&C Facility requires relative proximity to the railroad, road, the VAB and Launch Complex, yet provide sufficient isolation from normally populated areas. A suggested location is shown in Figure 7-2.

2. Alkali Metal Handling Facility

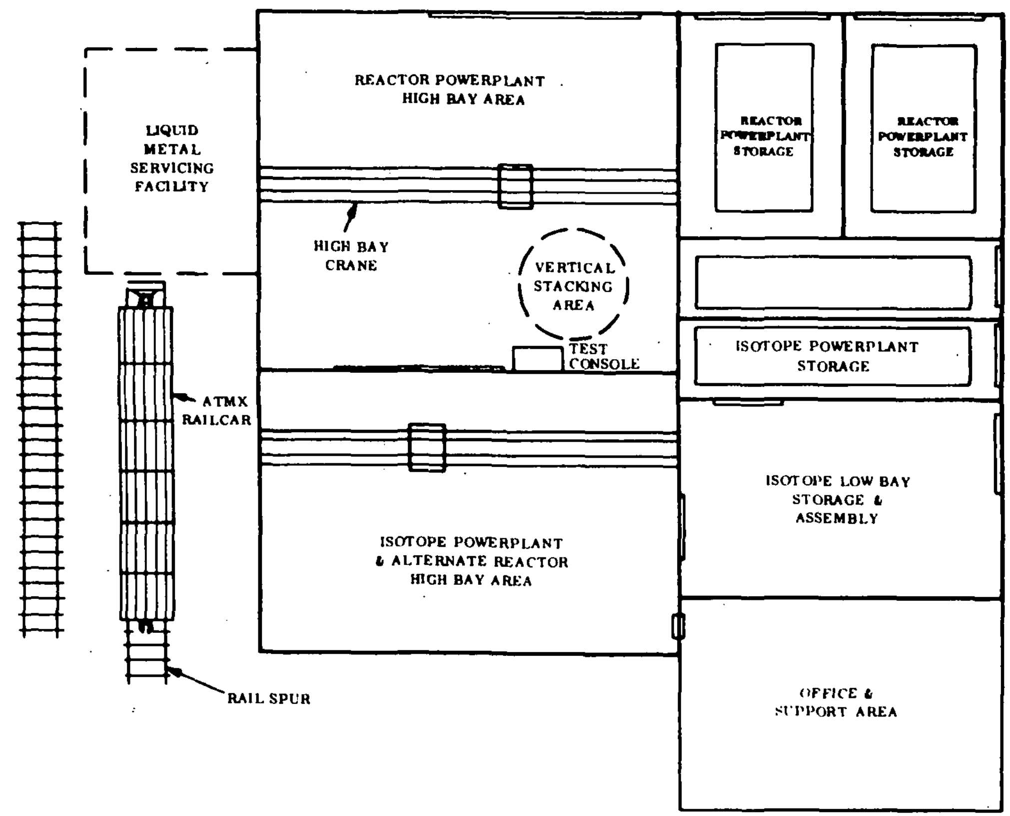
This facility provides for safe handling of NaK cooled NEP Stage power subsystem in the event of a liquid metal leak. The reactor power subsystem will be shipped direct from the factory with a full complement of NaK. NaK loops will remain filled and unopened throughout the remaining portion of the mission. This procedure eliminates the need for extensive liquid metal processing and charging facilities at KSC, but a limited servicing capability is still required to perform safing operations if liquid metal leaks or line ruptures should occur. After safing and cleanup and power module would be shipped back to the factory for repair.

This mode of operation appears to be appropriate for limited nuclear operations at KSC. However, a full capability liquid metal servicing facility should be considered when future multiple mission requirements dictate.

The Alkali Metal Handling Facility depicted in Figure 7-3 is typical of the limited facility which would be required, which is capable of expansion as requirements dictate. The preferred location is approximately 100 m from the Nuclear Storage and Checkout Facility, but within the same perimeter fence. An alternate location, providing greater accessibility, would be immediately adjacent to the NS&C Facility separated by fireproof walls.



NUCLEAR ASSEMBLY BUILDING



PRELIMINARY REQUIREMENTS

High Bay Area #1 (Prime Reactor System C/O & Vertical Stacking)	
Area	590m^2 (6350 ft^2)
Height (vertical stacking)	22m (72 ft)
(horizontal)	13m (45 ft)
Crane Capacity	46t (100Klb)
Door Size	12m Wide x 12m High
High Bay Area #2 (Prime Isotope System C/O & Secondary EPS Area)	
Area	440m^2 (4750 ft^2)
Height	12m (40 ft)
Crane Capacity	46t (100Klb)
Door Size	12m Wide x 11m High
Storage Area #1 (2 Bays for Reactor Storage)	
Area	330m^2 (3600 ft^2)
Height	9.2m (30 ft)
Storage Area #2 (Isotope Storage)	
Area	340m^2 (3650 ft^2)
Height	7m (23 ft)

Additional Requirements

- Office Space
- Environmental Control & Air Filtration System
- Helium-Zenon Servicing
- Inert Cover Gas Service (Argon, He, or N_2)
- Isotope Cooling
- Railroad Spur
- Controlled Access
- Multiple Escape Provisions
- Radiation alarm & detection system
- Fire Protection
- Liquid Metal Leak Detection

Figure 7-1. Nuclear Assembly and Storage Building

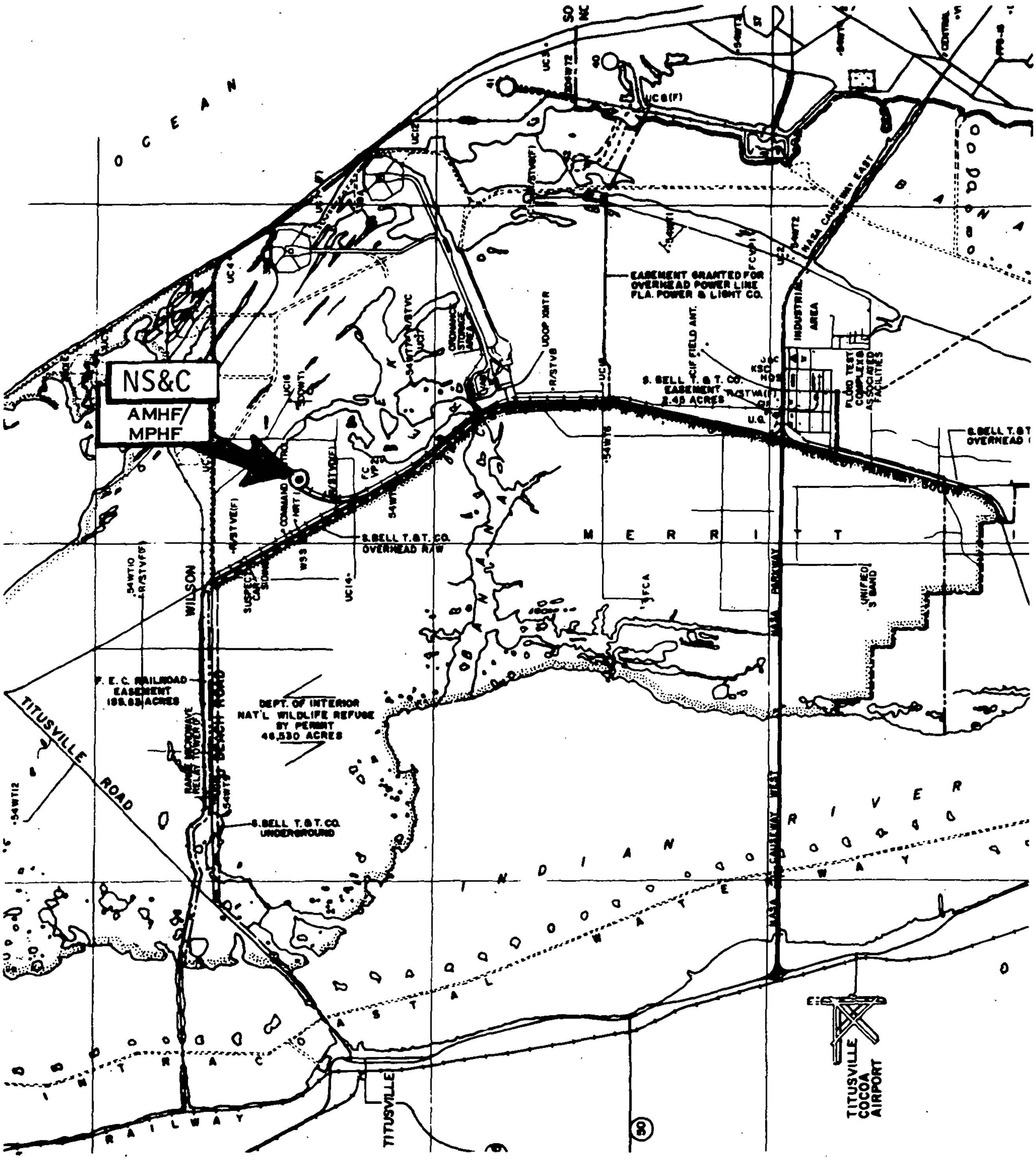
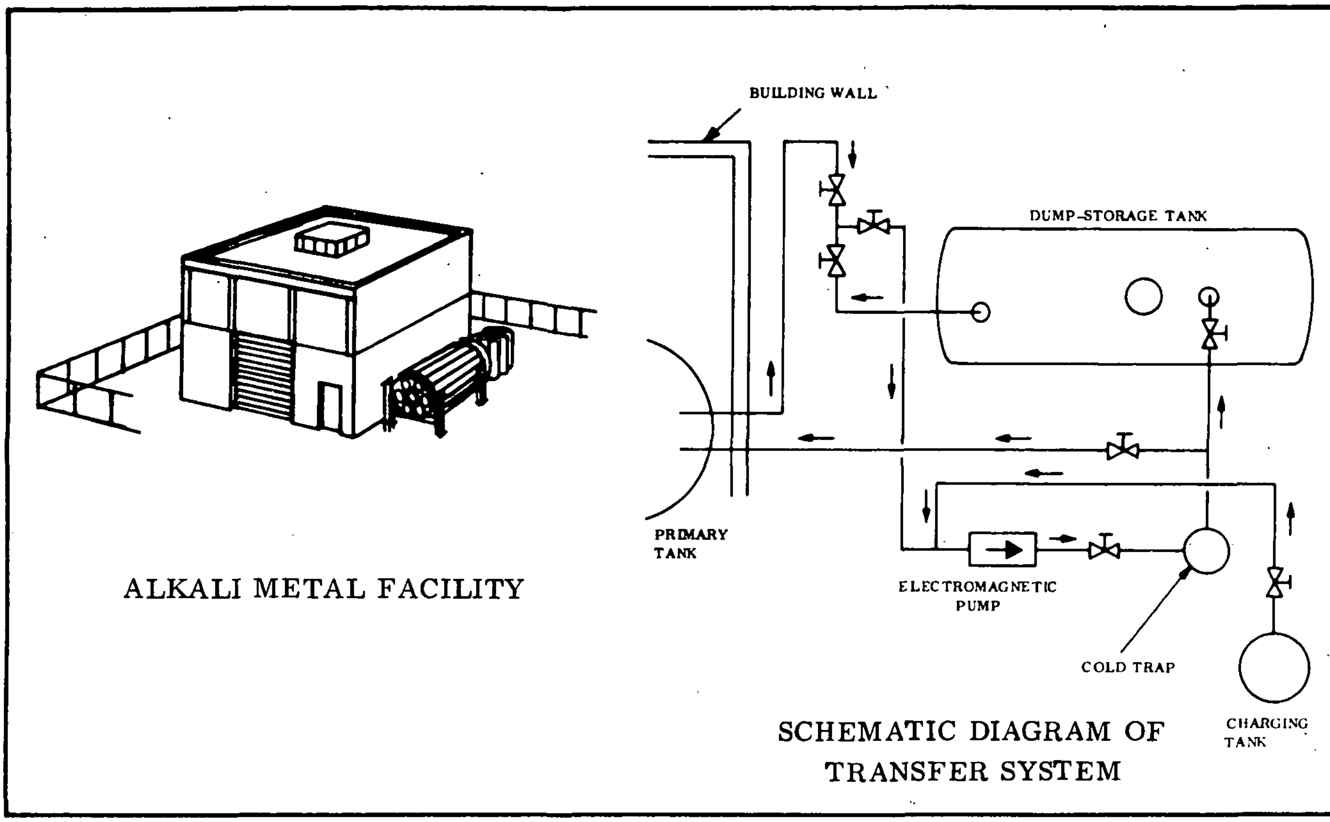


Figure 7-2. Suggested Location for Nuclear Storage and Checkout Facility



INITIAL FACILITY PROVISIONS

- FIRE PROTECTION
- PERSONNEL PROTECTION
- COVER GAS SUPPLY
- COVER GAS SAMPLING
- GAS PURIFICATION
- NaK BULK STORAGE
- LEAK TESTING
- PURGING
- EVACUATING/UNLOADING
- MINOR MAINTENANCE
- POST OPERATIVE CLEANING
- WASTE DISPOSAL

COMPLETE FACILITY PROVISIONS

- FIRE PROTECTION
- PERSONNEL PROTECTION
- COVER GAS SUPPLY
- COVER GAS SAMPLING
- GAS PURIFICATION
- LEAK TESTING
- PURGING
- EVACUATING/UNLOADING
- POST OPERATIVE CLEANING
- WASTE DISPOSAL

* BULK & TANK CAR STORAGE OF LIQUID METALS
* MATERIAL EXPOSURE FACILITIES
* LIQUID METAL SAMPLING
* LIQUID METAL PURIFICATION
* CHARGING SYSTEM
* SUPPLY SYSTEM
* MAJOR MAINTENANCE

*ADDITIONAL PROVISIONS

Figure 7-3. Alkali Metal Handling Facility

The prime considerations in the safe operation and design of the facility are the provisions for complete isolation from moisture and reactant substances along with proper fire protection. Liquid metal containers must be raised off the floor on blocks or grates to allow visual checks for leaks and corrosion. Drip pans are also required to catch and keep dripping metal off the concrete floor. Cover gases (helium, nitrogen, argon) should also be considered.

In all operations involving the use of liquid metals and nuclear hardware, it is vitally important that:

1. Cleanliness be maintained
2. Proper clothing is worn
3. "Buddy system" rules are rigidly enforced

Nuclear and fire safety precautions must be provided to protect workers, hardware and the surrounding environment. Radiation protection requirements can be met by providing shielded and isolated work and storage areas equipment with radiation detection monitoring and alarm instrumentation. Multiple access and escape routes must be planned.

Minimizing moisture within the building should be a design objective. The building should be waterproof and there should be no sprinkler system, exposed water pipes or steam lines in the work and storage areas. The floor should be sealed concrete sufficiently elevated to prevent water from entering. Continuously operating power ventilators with proper filtering should be provided to remove moisture. Smoking, eating and open flames should be prohibited in most areas. Switches, lights and motors must be explosion and arc-proof. When possible, cover gases should be maintained to further reduce any possible reactivity and exposure to the atmosphere.

3. Mercury Propellant Handling Facility

This facility provides for storage and handling of the NEP Stage mercury propellant, and for fueling the NEP Stage and the PLD prior to launch. This building can be very similar to the Alkali Metal Handling Facility.

4. Shipping Container

This container provides an inert, controlled environment for shipping the NEP Stage to the launch site. The shipping container must be equipped to monitor radiation, humidity, temperature and pressure and must provide the necessary inert cover gas environment, fire protection, alarms and warnings. This same transporter would be used for transport by airplane, barge, rail and roadway. It would also serve as the storage container and provide accessibility for checkout and component assembly. This vehicle may already be available on site for other requirements. A somewhat similar device has been successfully used by NASA in transporting, handling and storing the Nimbus spacecraft from the point of manufacture to the launch complex. The Air Force employs a similar technique in the transporting and handling of the operational Minuteman missiles. The reduced handling and increased environmental protection possible with the transporter concept provides significant safety advantages.

5. Handling Equipment

This equipment is necessary to remove the NEP Stage from the shipping container and, after completion of acceptance and checkout tests, to load the NEP Stage, installed in the transfer module, into the Shuttle cargo bay.

6. Transporter

A vehicle to transport the NEP Stage (plus Centaur or PLD) in its shipping container from the Nuclear Storage and Checkout Facility to the VAB where the payload and transfer Module are installed in the Shuttle cargo bay. This vehicle may be already available on site for other requirements.

Additional Ground Support Equipment that have been identified for NEP Stage operations are listed in Table 7-1.

Table 7-1. Facility and GSE Requirements

<u>Fabrication and Test</u>
TFE Test Equipment
Leak Test and Weld Inspection Equipment
NaK Charging and Purification Facility
Hot Test Facilities
Avionics Subsystem Simulator(s)
Low Voltage Electric Power Source
High Voltage Electric Power Source
Test Facility for Ion Engine Array Performance Test
Ion Engine Electrical Load Simulator
Propulsion System Simulator for Avionics Subsystem Test
Test Facility and Equipment for NEP Stage Test
Handling Rigs and Transporters for Each Subsystem
Shipping/Storage Containers with Environmental Control Packages for each Susbystem
<u>Arrival at Launch Site and Prelaunch</u>
Checkout Equipment for NEP Spacecraft Systems
Checkout Equipment for Centaur Systems
LOX and LH Fueling Facilities for Centaur
Inert Gas Supply and Handling Facilities
<u>Launch-Mission Completion</u>
Equipment to Monitor, Store and Process NEP Stage Information
Communications Equipment
Radio Tracking Capability

Specialized equipment and facilities are required during the NEP Stage fabrication and test operations. The individual TFE's will be back emission tested before assembly in the reactor. Equipment for weld inspection and leak checking of the power subsystem components is needed along with hot test facilities. Spacious facilities are required for both the nuclear testing of the power subsystem and the performance test of the ion engine array.

Specialized testing equipment is required for many of the assembly, subsystem and system acceptance tests. Electronic components which simulate various functions of the Avionics subsystem, such as, reactor control, pump control, ion engine control, etc., are needed. Electrical power sources which duplicate the low voltage output of the reactor and the high voltage output of the power conditioning assembly must be available. An ion engine electrical load simulator is needed for the propulsion system test while a simulator that duplicates the demands and responses of the propulsion system is required for the avionics subsystem acceptance test.

Each stage of fabrication and test requires specialized handling jigs, tools and transporters. Shipping/storage containers, some with attached environmental control devices, will be tailored to the individual size and weight requirements of each assembly, subsystem, etc.

At the launch site, test equipment is required which can check the operability of each of the functional subsystems in the NEP Stage (and the Centaur propulsive stage for interplanetary mission and the PLD for geocentric orbit missions).

At the launch pad, special facilities are required to fuel the Centaur stage (if the mission requires) which will be stowed in the Space Shuttle orbiter cargo bay. An inert gas facility may be needed to purge and flood the cargo bay of the Space Shuttle orbiter so that NEP Stage components can be activated while enclosed with the fueled Centaur.

During the flight stages of the NEP Stage, communications equipment, and data storage and processing equipment are required to monitor and evaluate the progress of the mission. On interplanetary missions, primary navigation is performed by the radio tracking facilities of the Deep Space Network, aided during the panet/comet intercept period by an on-board detector unit. All identified GSE (except for Centaur checkout equipment and facilities) are required for the NEP Stage, whether the mission is that of a Geosynchronous Orbit tug or an Interplanetary Multi-Mission Spacecraft.

7.2 OPERATIONAL EQUIPMENT

The key operational equipment identified are:

1. NEP Stage Transfer Module

   An adapter structure that facilitates handling, increases the safety of operations involving the NEP Stage, and minimizes the integration of the NEP Stage (or other payloads) with the Space Shuttle. This structure design mates with the load bearing attachment points in the Shuttle cargo bay, and with the load bearing attachment points on the NEP Stage. It must also be compatible with the Shuttle payload deployment mechanism. The use of the transfer module concept eliminates the requirement that the Shuttle cargo bay be designed specifically for the NEP Stage, or any other payload. The requirement for the transfer module concept is common to all NEP missions, interplanetary exploration or geosynchronous orbit applications.

2. Chemical Tug - Synchronous Payload Transfer Module

   The baseline geocentric orbit mission will require a similar transfer module to facilitate installation of the Chemical Tug, and its attached synchronous orbit payload, within the Shuttle cargo bay. The Chemical Tug then delivers this payload to the NEP Tug in the 14,800 km by 35,800 km parking orbit. This operational equipment is particular to the NEP Stage mission.

3. Propellant Logistic Depot (PLD)

   This hardware contains all the mercury propellant, and other consumables necessary to support the NEP Stage in orbit during its geocentric mission operational life. It is launched with the NEP Stage on the initial Shuttle launch and placed in the parking orbit by the NEP Stage. The PLD includes an attitude control system, a tracking beacon, and passive docking systems, in addition to the tankage required to contain the NEP Stage support consumables. This operational equipment is particular to the NEP Stage mission.

4. Auxiliary Power Supply

An auxiliary power supply, required during launch and prelaunch activities, is attached to the transfer module. One function of the auxiliary power supply might be to provide electrical power to heaters to prevent NaK freeze-up during launch operations.

SECTION 8
DEVELOPMENT SCHEDULES AND COSTS

8.1 SUMMARY

This section presents the gross development schedules and costs for the NEP system. The multi-mission NEP Stage development costs, propulsion and avionics system, and recurring costs are defined.

Alternate propulsion system development schedules have been examined which illustrate the cost impact of alternate levels of technology, system prototype and/or complete NEP system ground tests. Extensive system and subsystem prototype tests do not appear to be required to assure a reasonably high probability of mission success. Because of the inherent reliability of the thermionic NEP system, combined nuclear system tests are not necessary in the development program, although such tests have been considered and their cost evaluated.

The primary purpose of this section is to define the NEP system development costs. The specific objectives are:

1. To provide gross NEP Stage development schedule and costs
2. To define costs for Ground Support Equipment (GSE), operational equipment, and mission operations
3. To provide visibility of program cost elements

The scope of the NEP system development program includes two main program options:

1. A baseline program which is designed to provide a 20,000-hour (full power) NEP system for early 1980's multi-mission applications. A high degree of success is assured with a moderate cost by employing a comprehensive technology development effort coupled with limited prototype tests of key NEP subsystems.

2. A minimum program which is designed to provide a 10,000-hour (full power NEP system for early 1980's multi-mission applications. Emphasis is placed primarily on technology development in order to minimize program costs. However, a moderate degree of success may be expected because of reduced NEP system full power life requirements, relative to the baseline program.

The cost impact of extensive use of beryllium structure is assessed. The recurring costs associated with the liquid metal heat rejection subsystem of the NEP power subsystem are investigated. Gross estimates are presented for the total NEP system recurring costs.

8.1.1 KEY GUIDELINES AND ASSUMPTIONS

Table 8-1 shows the key guidelines and assumptions on which the NEP system development schedules and costs are based. A 120 kWe end thrust NEP system employing an internal fuel thermionic reactor, assumed to deliver 40 Vdc, is employed as the baseline system for the NEP system development cost estimates. The system operating life objective is 50,000 hours, with 10,000 hours to 20,000 hours full power capability for the power subsystem.

Stainless steel is assumed for the power subsystem liquid metal containment. The main radiator is assumed to consist of sodium filled stainless steel heat pipes. (The cost impact of a beryllium radiator structure is also assessed.) The NEP system structure is assumed to be aluminum or stainless steel, depending on the temperature level. The impact on NEP system costs of extensive use of beryllium structure is assessed.

The baseline development program employs tests to demonstrate technology readiness of key components such as thermionic fuel elements. Liquid metal heat rejection loop components are considered relatively state-of-the-art because of extensive development completed in this area by Atomic International for the AEC and General Electric for NASA. However, limited component development is planned for this particular application. The solar electric program is assumed to provide the basic thrust system ion engine and power conditioning technology. Partial and limited full ion engine array tests are scheduled to verify the application of this technology to the NEP system.

Table 8-1. Key Guidelines and Assumptions

NEP SYSTEM

- 120 kWe TO THRUST SUBSYSTEM
- 40 VDC INTERNAL FUEL REACTOR
- STAINLESS STEEL LIQUID METAL CONTAINMENT
- END THRUST CONFIGURATION
- HEAT PIPE RADIATOR
- STAINLESS STEEL AND ALUMINUM STRUCTURE

DEVELOPMENT APPROACH

- DEMONSTRATE TECHNOLOGY READINESS BY COMPONENT TESTS
    - THERMIONIC FUEL ELEMENTS
    - EM PUMP
    - ION ENGINES
- SYSTEM OPERATIONAL TESTS
    - GROUND PROTOTYPE REACTOR
    - LIQUID METAL HEAT REJECTION (QUARTER RADIATOR)
    - ION ENGINE ARRAY/POWER CONDITIONING
- FLIGHT HARDWARE TYPE ACCEPTANCE (TA) NEP SYSTEM (DUMMY REACTOR)

SCHEDULES AND COSTS

- BUILD ONE FLIGHT NEP SYSTEM
- ESTIMATED GSE AND FACILITY COSTS
- REACTOR COSTS BASED ON GULF GENERAL ATOMIC DATA
- ION ENGINE COSTS BASED ON JET PROPULSION LABORATORY DATA
- SHIELD COSTS PER ATOMICS INTERNATIONAL DATA
- FISCAL 1972 DOLLARS
- FOUR MANPOWER CLASSES
    - ENGINEERING @ $11/HR
    - DRAFTING @ 7/HR
    - TECHNICIAN @ 8/HR
    - HOURLY @ 5/HR
- OVERHEAD AT 120 PERCENT OF TOTAL LABOR
- GENERAL AND ADMINISTRATIVE COSTS AT 10 PERCENT LOM
- NO FEE INCLUDED IN COSTS

- COMPLETE STRUCTURAL DEVELOPMENT HARDWARE
    - DYNAMIC MOCKUP
    - THERMAL MOCKUP
    - ENGINEERING DEVELOPMENT (FIT)
    - ELECTRICAL HARNESS
- NEP STAGE - KICK STAGE ADAPTER STRUCTURE

- COMPLETE TOOLING COSTS
- PROPELLANT SUBSYSTEM TANK DEVELOPMENT
- NO SHIELD TECHNOLOGY DEVELOPMENT
- NO NON-NUCLEAR INSTRUMENTATION TECHNOLOGY DEVELOPMENT
- NO SHUTTLE LAUNCH/CENTAUR COSTS INCLUDED

The baseline program employs system operational costs to verify the performance of major NEP systems. These include the thermionic reactor, the main heat rejection system employing one-quarter to one-third of the full size radiator, and the ion engine array with its associated power conditioning. The type acceptance tests for the flight NEP system employs a complete spacecraft, except that the reactor mechanical (mass) and electrical characteristics are simulated.

All development program options evaluated assume that completion includes the design, fabrication, and launch of one NEP system. All basic ground support equipment and facility costs identified are included. NEP system component development and flight system cost data have been obtained from NASA/JPL, the AEC, and their contractors where necessary.

All costs assume FY 1972 dollars. No escalation and no contingency costs are included. Four manpower classes are employed where applicable: engineering, drafting, technician and hourly. Assumed overhead is 120 percent of labor dollars. General and Administrative costs are assumed at 10 percent of total labor overhead and materials.

No fee is included in the costs presented. This will amount to 5 to 10 percent of the program total, depending on the contracting structure and the number of subcontractors employed.

No allowance is included for government agency monitoring and other participation. This could add an additional 8 to 10 percent to the total program cost. Alternately, performance of key program elements by government laboratories and agencies would act to reduce total program costs.

An extensive structural development program will be required for the NEP system. This includes dynamic and thermal mockups and tests, an engineering development mockup, and an electrical harness mockup. Costs are included for the adapter structure required to attach the NEP Stage to the high energy kick stage (i. e., Centaur) for interplanetary applications. Costs are also identified for the docking structure required for geocentric orbit applications.

Complete tooling costs are included as a part of the GSE. Development of the propellant tank, which also functions as the main gamma shielding, is assumed to be required for all NEP system development program options evaluated. It is doubtful that this technology can be taken directly from the Solar Electric Propulsion (SEP) program, because of the unique geometry and nuclear radiation environment operation required for the NEP system.

No shielding or non-nuclear instrumentation and control technology development requirements are identified.

Flight nuclear safety and other safety costs are organized in the Program Management and Systems Engineering tasks. However, total safety costs are often quoted separately for fiscal visibility. No reactor destructive nuclear safety test costs are included.

Identified Launch and Mission Operations costs are limited to contractor support, and are identical for both programs evaluated. No launch vehicle costs are included in the program total costs, although their costs are estimated.

8.1.2 BASELINE NEP SYSTEM PROGRAM

The summary schedule for the baseline NEP system development program is shown in Figure 8-1. The baseline program is assumed to begin in Fiscal Year 1973 and extend for eleven years to meet an early 1980's launch objective for a 20,000 full power hour life NEP system.

Key elements of the baseline program are:

1. Inclusion of two ground reactor tests, TREX and a Ground Prototype Reactor.
2. Strong dependence on SEP technology, although a partial ion engine array development test and a full ion engine array test are included to verify performance in the NEP configuration.
3. Early requirements for GSE, particularly structural simulation, and for facilities for reactor tests. The NEP system assembly test and checkout facility is required about three years before launch.

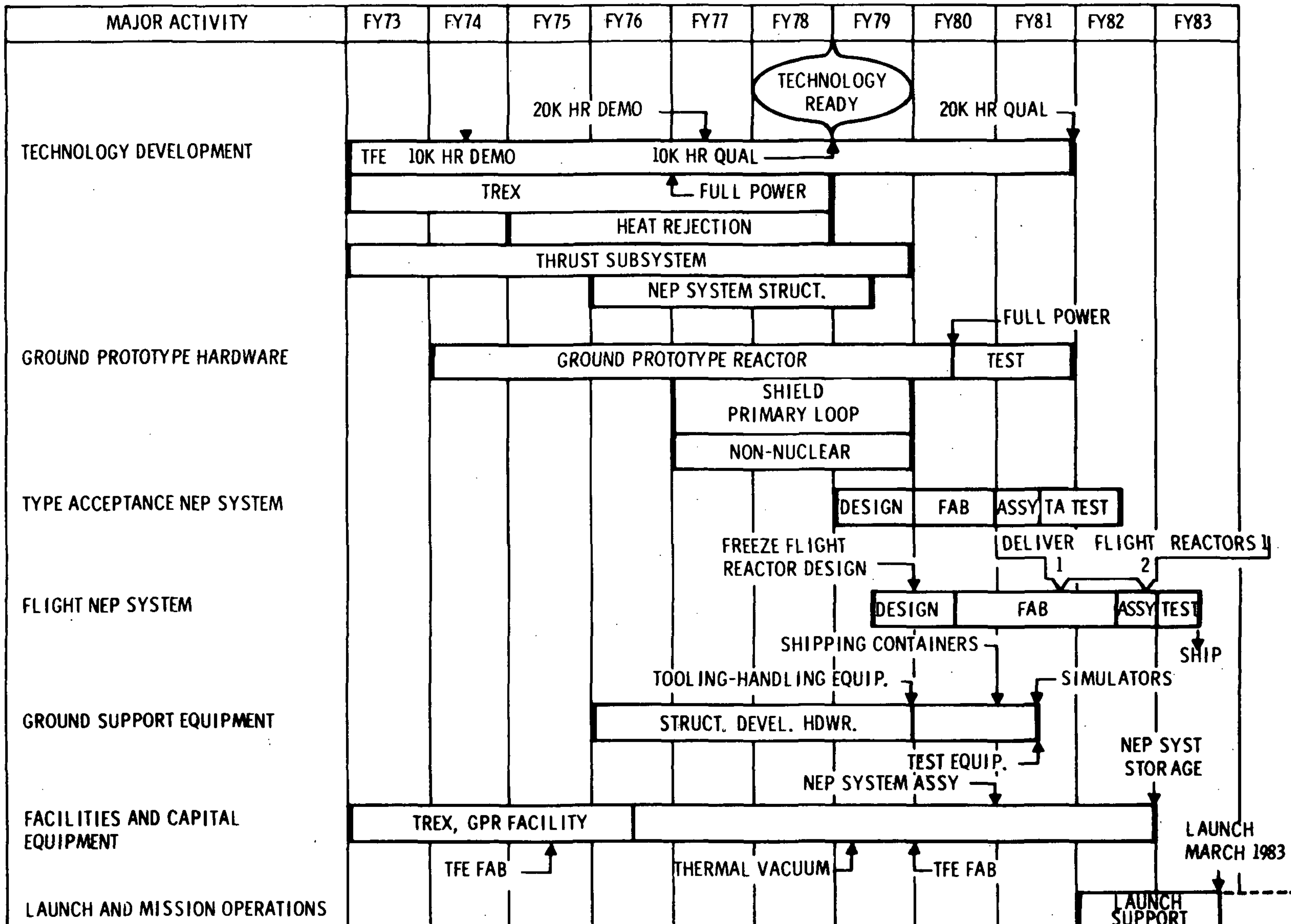


Figure 8-1. Summary Schedule for Baseline NEP System Program

4. A TFE design with proven continuous 20,000 life capability is qualified two years before launch.

5. Technology ready and preliminary mission approval occur in FY 1978 after demonstration of the feasibility of a 20,000-hour TFE life, and with the qualification of the 10,000-hour life TFE. The TA NEP system design is initiated.

Figure 8-2 shows the baseline NEP system development program cost elements grouped to present program costs in terms of basic development, the total flight program, and mission support. The \$160 M development program represents about 58 percent of the total. The \$113 M flight program cost is about 41 percent of the total. The contractor mission support function constitutes less than one percent of the program total. Required facilities will add \$35.4 M to these costs.

The breakdown of the baseline NEP system development program shown in Figure 8-3 emphasizes the cost elements of the NEP system hardware. The percent of the total program costs are also shown. Cost data particular to this chart are:

1. Non-nuclear instrumentation and controls
2. Thrust subsystem (exclusive of the propellant subsystem)
3. Propellant subsystem
4. Radiation shield
5. Power subsystem (exclusive of shield)
6. Structural development

Figure 8-4 presents program costs as a function of fiscal year for the \$275 M baseline NEP system development program. Peak costs of \$56 M are estimated for Fiscal Year 1979. These costs include overhead, G&A, and a total of approximately \$ 60 M in material costs. These costs are based on Fiscal 1972 dollars, and do <u>not</u> include any allowance for contingency, escalation, or U-235 fuel costs.

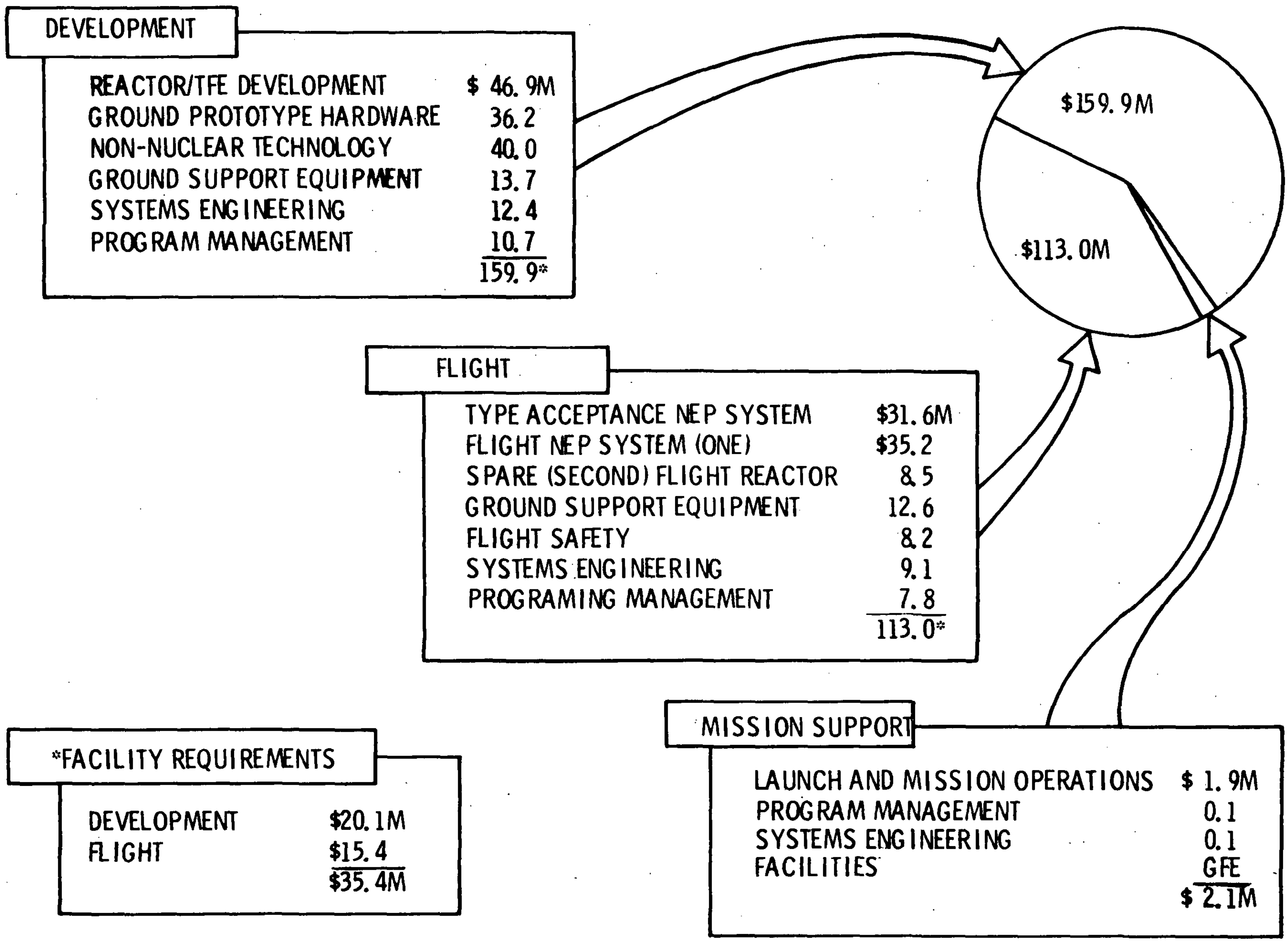


Figure 8-2. Cost Summary for Baseline NEP System Program

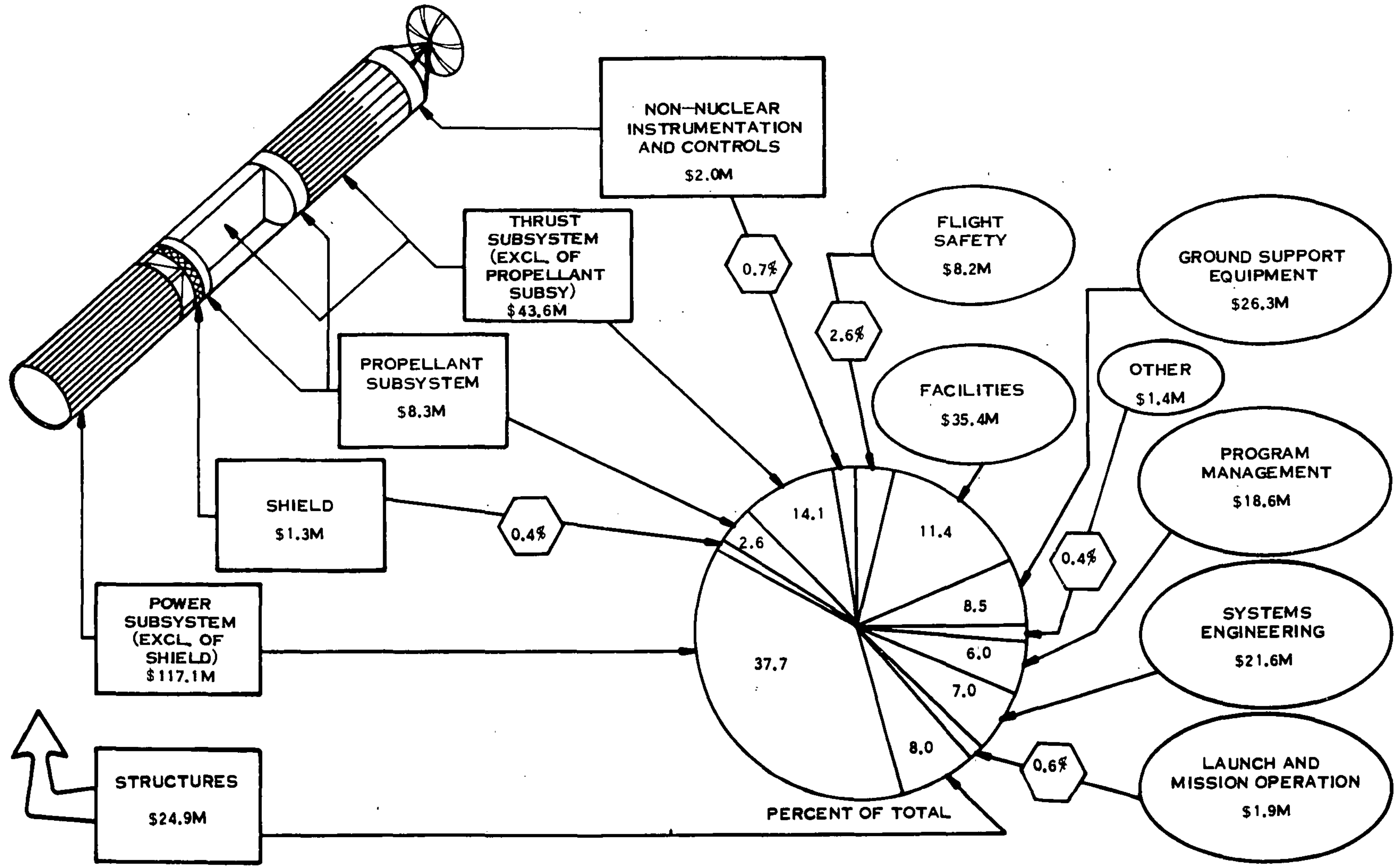


Figure 8-3. Key Cost Elements for Baseline NEP System Program

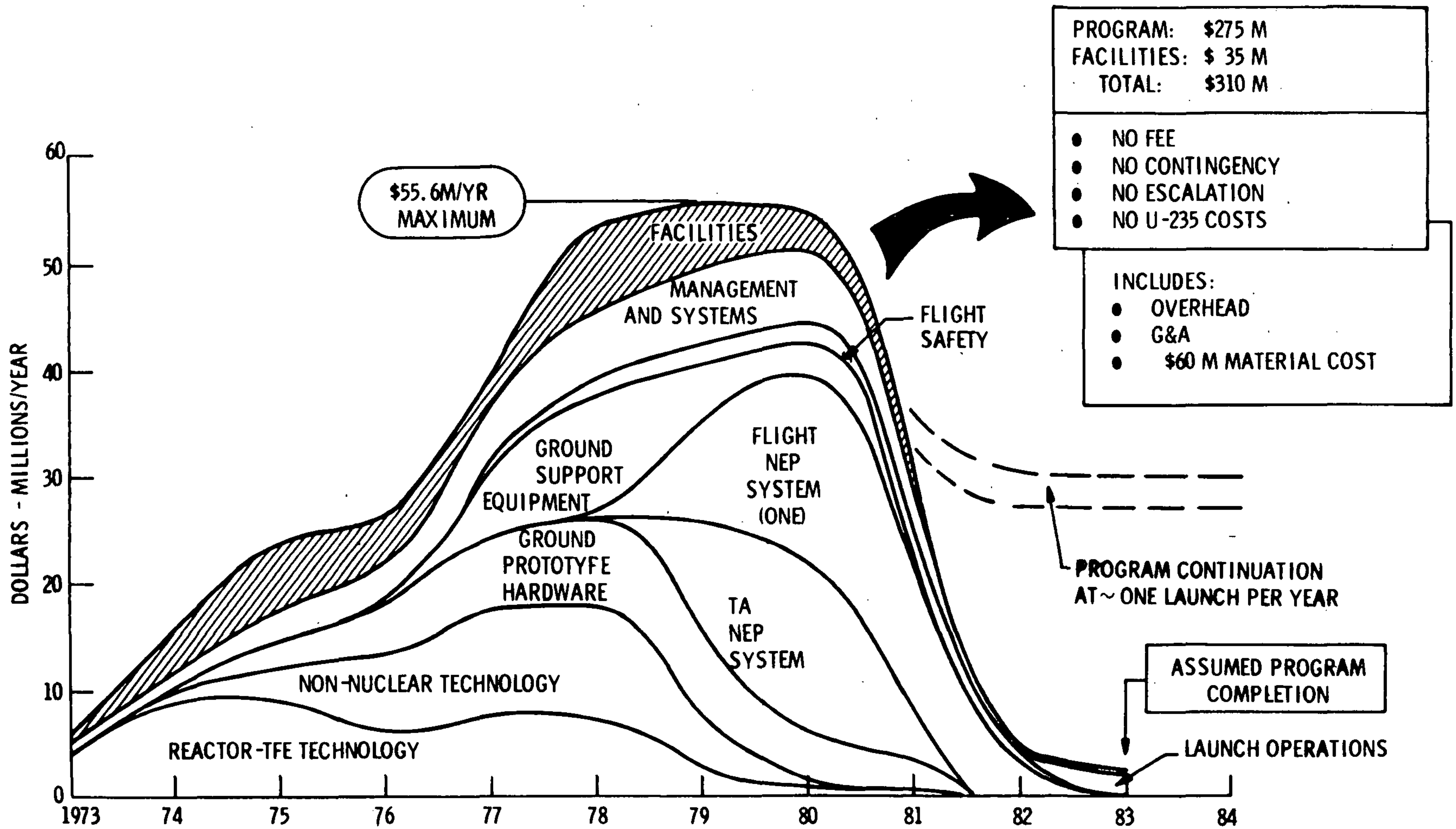


Figure 8-4. Baseline NEP System Program Total Dollars by Fiscal Year

The contribution of the major task elements are presented. There is a clear flow of funding from technology, to ground prototype hardware, to the TA NEP system, and to the Flight NEP systems. The early requirement for facilities and GSE and their impact on annual program funding requirements is clearly indicated.

Total dollars for key program elements as a function of fiscal year are presented in Table 8-2. Key program milestones are indicated. The baseline NEP system development program incorporates a $27 M TFE development program. The total cost for the two test reactors, including test operation, is $ 48 M. Other technology development, including structures and ion engine array, accounts for $48 M. These totals do not include related program management and safety.

Total TA and Flight NEP Systems costs are $76 M. Flight safety costs are about $8 M. Management and Systems Engineering are $42 M (Launch and Mission Operations are included at $2 M). Total GSE costs are $26 M. Facility costs add $35 M to the $275 M NEP System program.

8.1.3 MINIMUM NEP SYSTEM PROGRAM

The summary schedule for the minimum NEP system development program is presented in Figure 8-5. This minimum program is assumed to begin in Fiscal Year 1973 and extend for eleven years to meet an early 1980's launch objective for a 10,000 full power hour life NEP system.

Key elements of this minimum program are:

1. Program costs are minimized during the first five years.
2. Only one ground reactor test is included.
3. Major dependence on SEP technology. The only development included for the thrust and propellant systems are for the integrated propellant-shield tank, power conditioning nuclear environment tests, and a partial array ion engine test.

Table 8-2. Total Dollars by Key Program Elements Baseline NEP System Program

PROGRAM ELEMENT			FISCAL YEAR 73	74	75	76	77	78	79	80	81	82	83	TOTALS
POWER SUBSYSTEM DEVELOPMENT														
	TFE DEVELOPMENT AND DEMONSTRATION		3	4	4	4	4	4	2	1	1			$27M
	REACTORS	TREX AND TEST OPERATIONS	1	5	4	3	3	4						20
		GPR AND TEST OPERATIONS			3	4	5	4	5	4	3			28
	OTHER POWER SUBSYSTEM TECHNOLOGY				1	3	3	2	2					11
THRUST SUBSYSTEM DEVELOPMENT				1	2	5	9	12	7	1				37
				10,000 HR TFE DEMO			20,000 HR TFE DEMO	10,000 HR TFE QUAL	TECHNOLOGY READY	GPR AT FULL POWER	20,000 HR TFE QUAL TA NEP SYSTEM	FLIGHT NEP SYSTEM	LAUNCH	
TYPE ACCEPTANCE AND FLIGHT NEP SYSTEMS								1	20	33	19	3		76
FLIGHT SAFETY							1	1	2	1	1	1	1	8
MANAGEMENT, SYSTEM AND LAUNCH			1	2	2	3	5	7	7	7	4	2	2	42
GROUND SUPPORT EQUIPMENT						1	5	11	5	3	1			26
		TOTALS	5	12	16	23	35	46	50	50	29	6	3	$275M
FACILITIES			1	3	7	3	3	7	6	3	1	1		35
		TOTALS	6	15	23	26	38	53	56	53	30	7	3	$310M

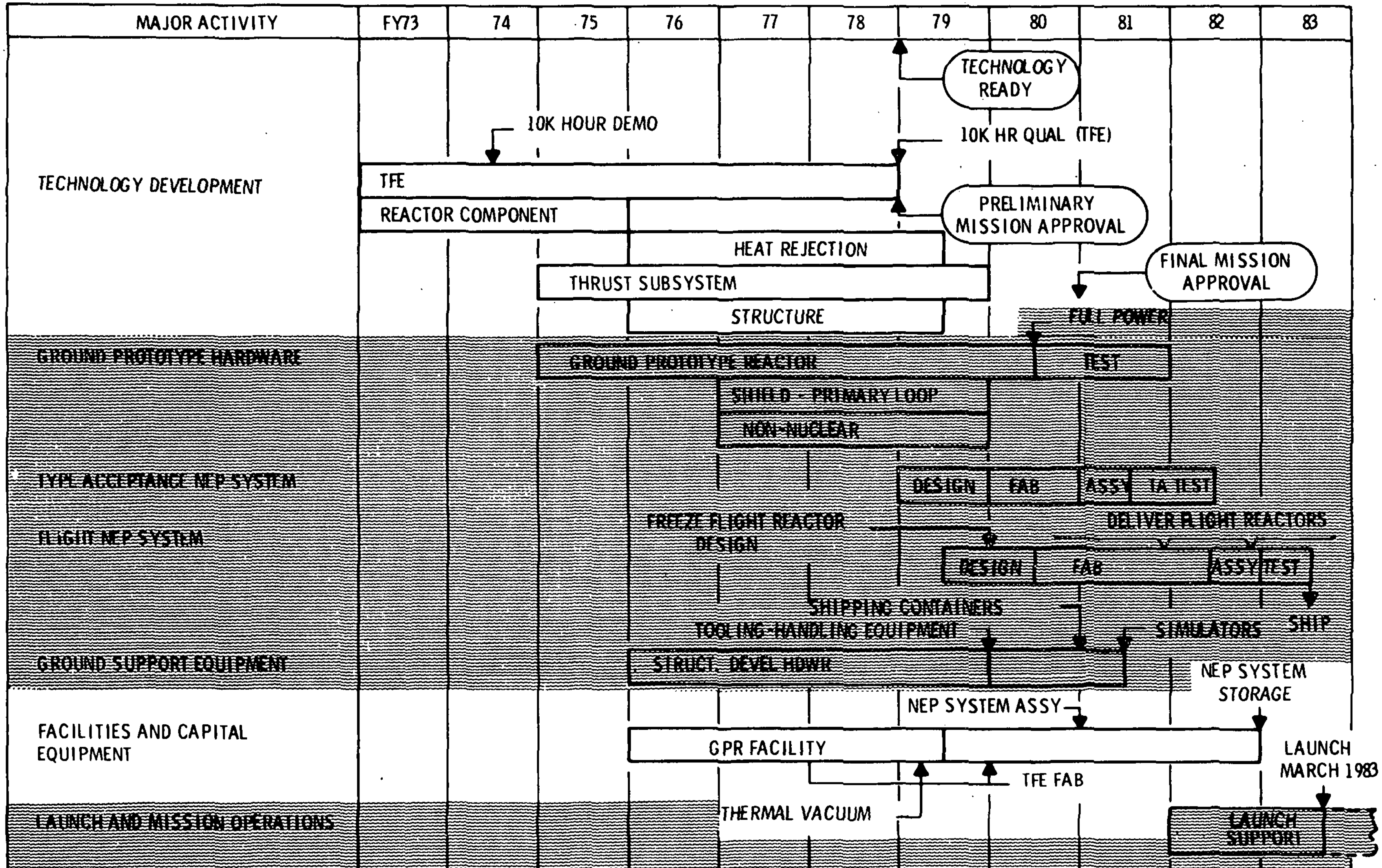


Figure 8-5. Summary Schedule Minimum NEP System Program

4. Early requirements for GSE, particularly structural simulators, remain in common with the Baseline program. Facility requirements are almost identical, except that no TREX facility is required. The schedule for facility availability is delayed one to two years, relative to the Baseline Program.

5. Technology readiness and final mission approval occur in Fiscal Year 1978 with the demonstration of continuous 10,000-hour TFE life capability.

Figure 8-6 shows the minimum NEP system development program cost elements grouped to present program costs in terms of basic development, the total flight program and mission support. The $118 M development program represents about 50 percent of the total. The $113 M flight program cost is about 49 percent of the total. The major change, relative to the baseline program, is a $42 M decrease in the development program. The contractor mission support function constitutes less than one percent of the program total.

Figure 8-7 presents a breakdown of the minimum NEP system development program emphasizing the cost elements of the NEP system hardware. The percent of the total program costs are also shown.

Program costs are presented in Figure 8-8 as a function of fiscal year for the $233 M minimum NEP system development program. Peak costs of $58.3 M are estimated for Fiscal 1979. These costs include overhead, G&A, and a total of approximately $50 M in material costs. These costs are based on Fiscal 1972 dollars, and do not include any allowance for contingency, escalation, or U-235 fuel costs. No fee is included in these costs.

Total dollars for key program elements are presented in Table 8-3 as a function of fiscal year. Key program milestones are indicated. The minimum NEP system development program incorporates a $23 M TFE development program. The total cost for the test reactor, including test operations, is $28 M. Other technology development, including structures, accounts for $40 M. These totals do not include related program management and safety.

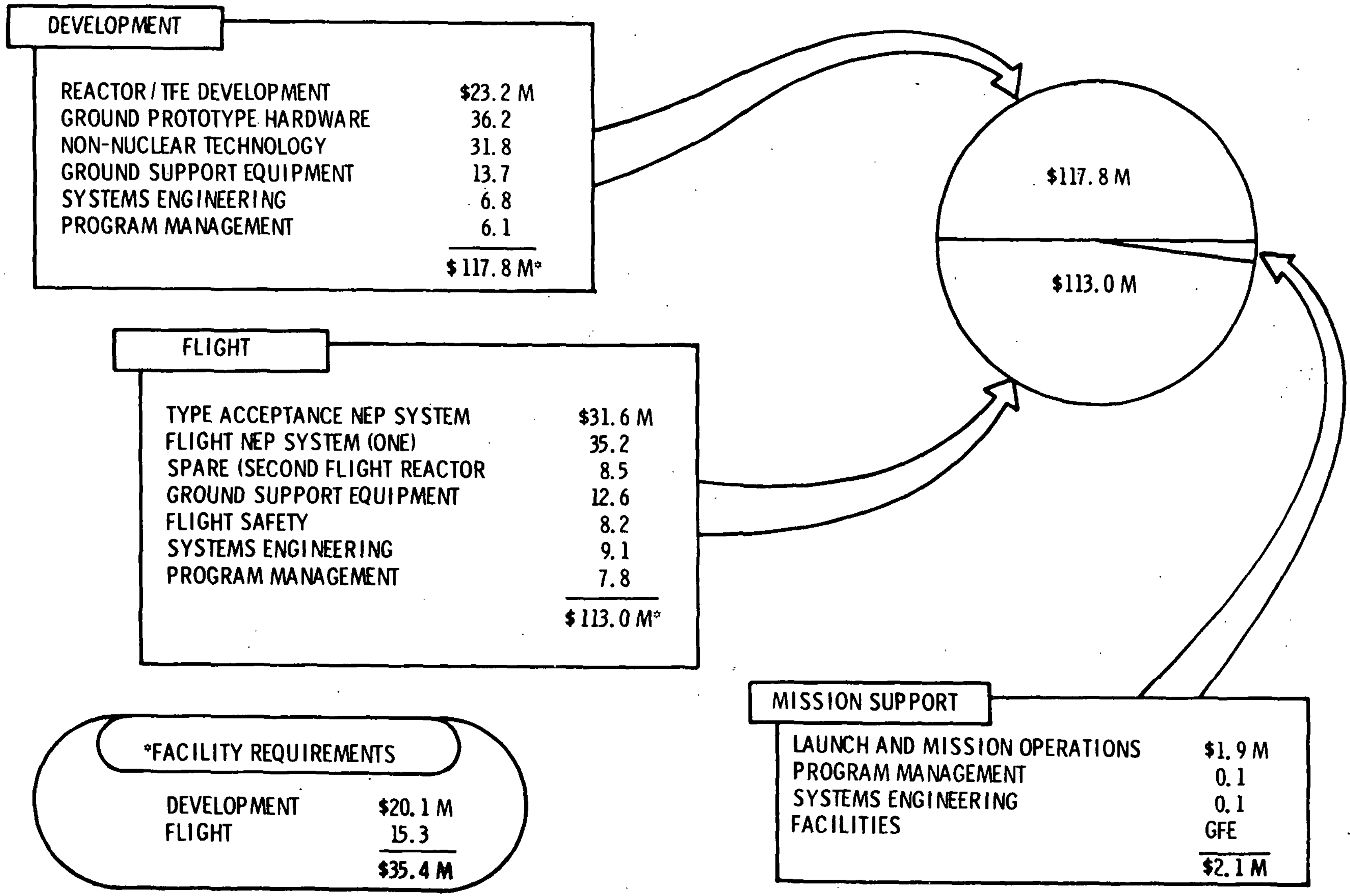


Figure 8-6. Cost Summary for Minimum NEP System Program

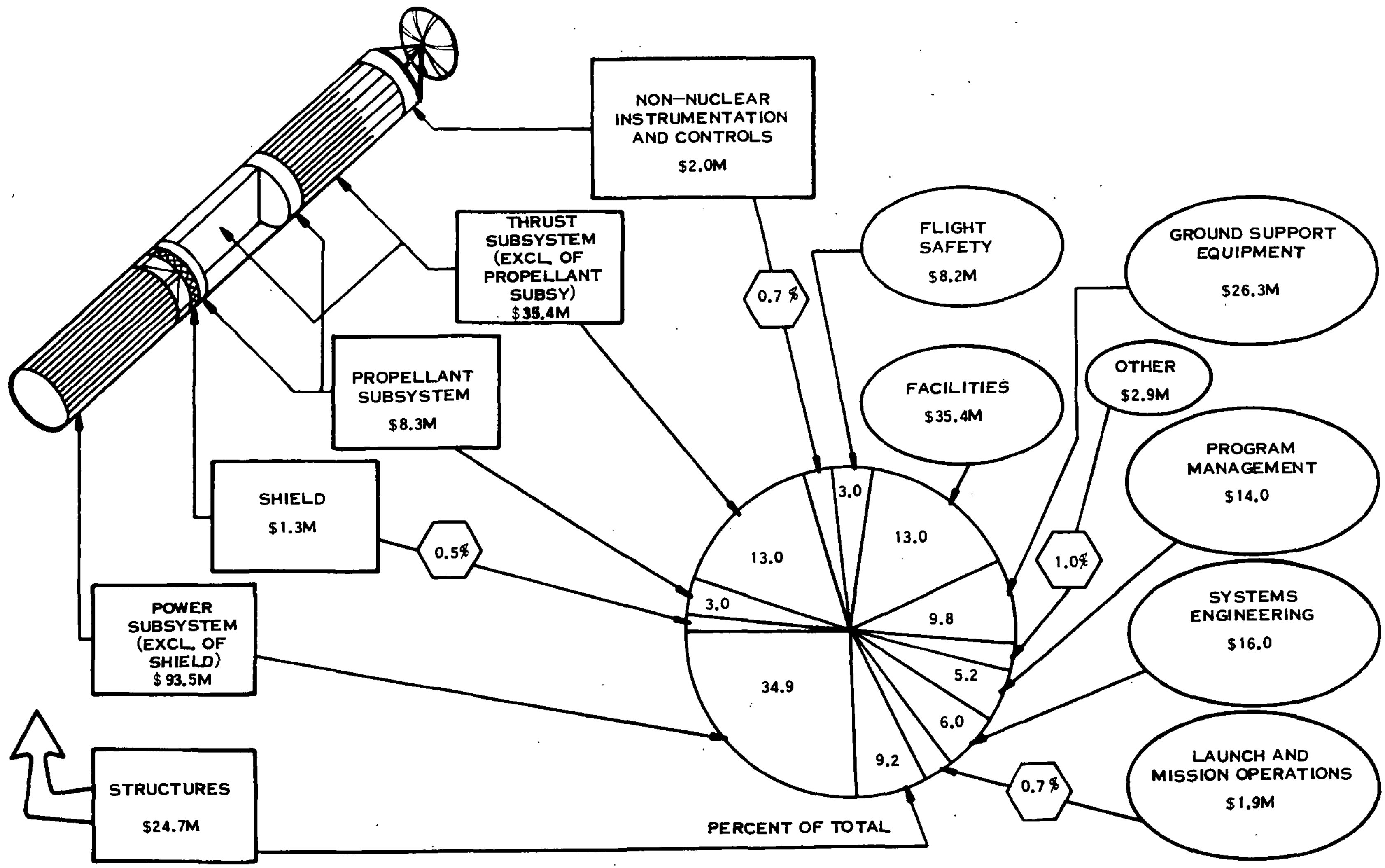


Figure 8-7. Key Cost Elements for Minimum NEP System Program

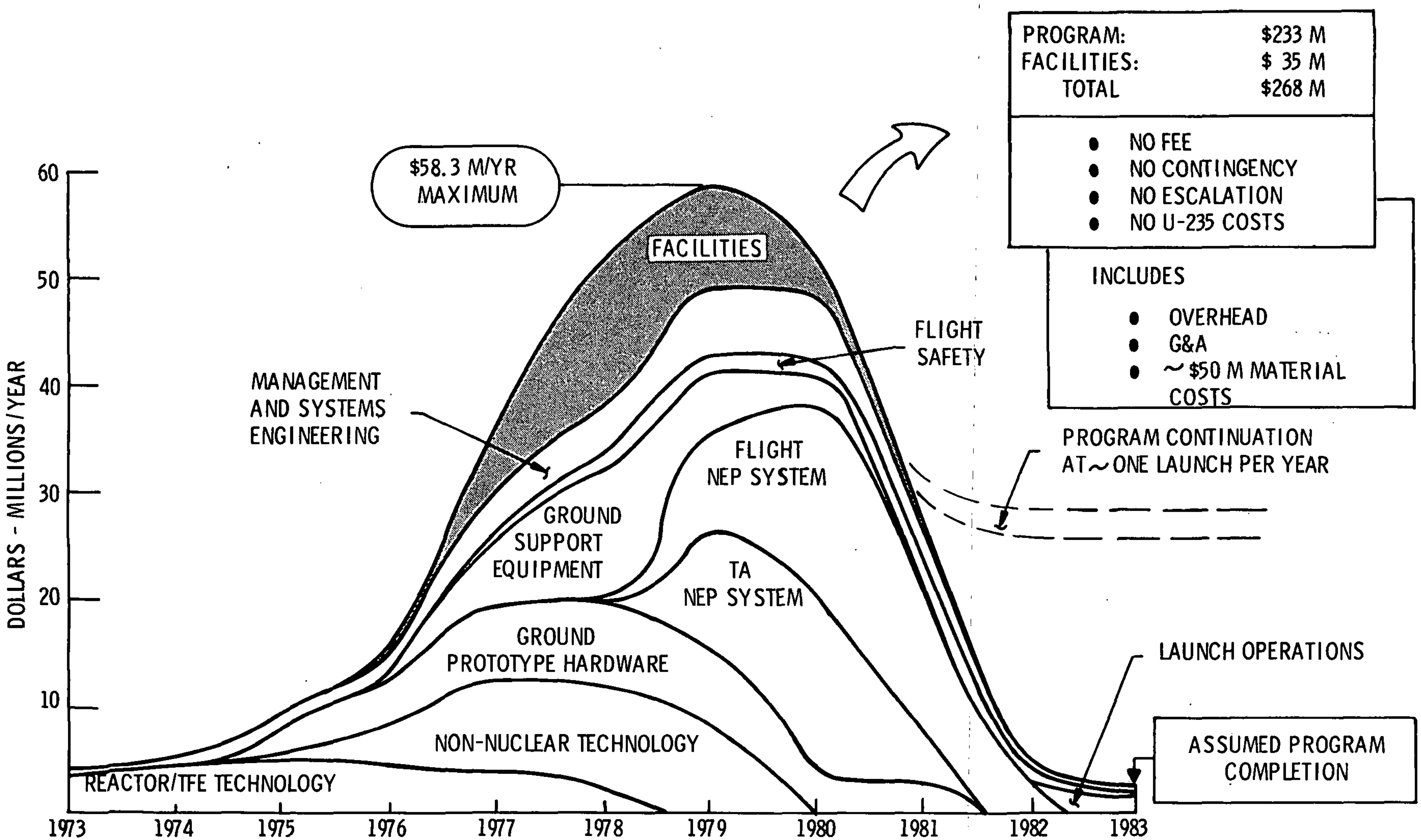
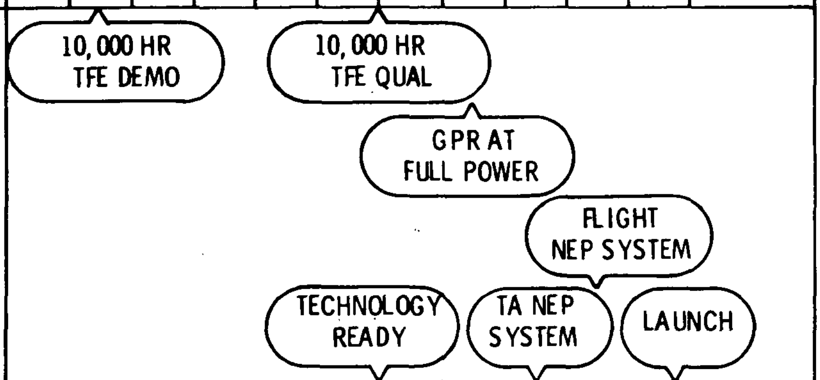


Figure 8-8. Minimum NEP System Program Total Dollars by Fiscal Year

Table 8-3. Total Dollars by Key Program Elements
Minimum NEP System Program

PROGRAM ELEMENT		FISCAL YEAR 73	74	75	76	77	78	79	80	81	82	83	TOTALS
POWER SUBSYSTEM DEVELOPMENT													
	TFE DEVELOPMENT AND DEMONSTRATION	4	4	5	4	4	2	-	-	-	-	-	$ 23M
	REACTOR – GPR AND TEST OPERATIONS			3	4	5	4	5	4	3			28
	OTHER POWER SUBSYSTEM TECHNOLOGY				2	3	4	2					11
THRUST SUBSYSTEM DEVELOPMENT				1	2	7	10	9					29
TYPE ACCEPTANCE AND FLIGHT NEP SYSTEMS							1	20	33	19	3		76
FLIGHT SAFETY						1	1	2	1	1	1	1	8
MANAGEMENT, SYSTEMS AND LAUNCH		1	1	1	2	4	4	6	6	3	2	2	32
GROUND SUPPORT EQUIPMENT					1	5	11	5	3	1			26
	TOTALS	5	5	10	15	29	37	49	47	27	6	3	$233M
FACILITIES					1	6	14	9	3	2			35
	TOTALS	5	5	10	16	35	51	58	50	29	6	3	$268M



Total TA and Flight NEP Systems costs are \$76M. Flight safety costs are about \$8M. Management and System Engineering are \$32M (Launch and Mission Operations are included at \$2M). Total GSE and Facility costs are \$61M.

Key programmatic options and schedule milestones are compared in Table 8-4 for the NEP system development program alternates investigated. There alternates employ an extensive TFE development program and a Ground Prototype (flight configured) reactor test. In addition, the baseline development program includes an earlier TREX reactor, which is not necessarily flight-configured.

Table 8-4. Particular Guidelines and Constraints
NEP System Development Options

	TFE DEVELOPMENT	TREX	GROUND PROTOTYPE REACTOR TEST	ION ENGINE TESTS	PROTOTYPE ION ENGINE ARRAY LIFE TEST	TECHNOLOGY READY	PRELIMINARY MISSION APPROVAL	FINAL MISSION APPROVAL	10,000 HOUR TFE QUALIFICATION	20,000 HOUR TFE QUALIFICATION	LAUNCH
BASELINE PROGRAM 20,000 FULL POWER HOUR MISSION	Y	Y	Y	Y	Y	FY 78	FY 78	FY 80	FY 78	FY 81	FY 83
MINIMUM PROGRAM 10,000 FULL POWER HOUR MISSION	Y	N	Y	Y	N	FY 78	FY 78	FY 78	FY 78	N	FY 83

Y = YES – INCLUDED

N = NOT INCLUDED

The minimum program depends significantly on the SEP program for the thrust subsystem technology. Partial ion engine tests are included, and limited power conditioning nuclear environment tests are scheduled. The baseline program includes one partial ion array test and one full ion engine array test, utilizing SEP technology configured for the NEP system.

The baseline program is assumed to be technology ready in FY 78, with the demonstration of continuous 10,000 hour TFE full power life capability, and potential for a similar 20,000 hour life. Mission approval follows in FY 80. Mission approval and technology readiness for the minimum program occurs in Fiscal 1978, because its mission life objective is assumed at 10,000 full power hours.

Table 8-5 compares the major cost elements of the two NEP system development program alternates. The Program Management and Systems Engineering functions are seen to be a fairly constant percent of the totals at 6 to 7 percent, and 7 to 8 percent, respectively. Flight safety is 3 to 4 percent for the baseline and minimum programs. Ground Prototype Hardware test percentages vary from 13 to 16 percent.

Total dollar values are constant for TA and Flight NEP Systems for both programs. Launch and Mission Operations also show constant dollars as do GSE total dollars and total facility dollars.

8.2 PROPULSION SYSTEM COSTS

The gross propulsion system development costs are detailed in this section. Figure 8-9 shows the top level work breakdown structure for the NEP System development program. Indicated on this chart are the subsections which contain the detailed costing of the respective items.

8.2.1 DEVELOPMENT PROGRAM

The baseline NEP System development program objective is to provide a NEP system with a 20,000 hour full power capability for an early 1980's mission. This NEP system would perform all identified interplanetary missions, as well as geocentric earth orbital missions. The approach employed is a high level of technology development effort coupled with prototype tests of the major NEP systems, the thrust subsystem, and major elements of the power subsystem.

Table 8-5. Dollar Summary - NEP System Development Program Comparison

	BASELINE PROGRAM		MINIMUM PROGRAM	
	$M	PERCENT	$M	PERCENT
PROGRAM MANAGEMENT	$ 18.6	6.7%	$ 14.0	6.0%
SYSTEMS ENGINEERING	21.6	7.8	16.0	7.0
FLIGHT SAFETY	8.2	3.0	8.2	3.5
TECHNOLOGY DEVELOPMENT	86.9	31.7	55.1	23.6
GROUND PROTOTYPE HARDWARE	36.2	13.2	36.2	15.5
TYPE ACCEPTANCE NEP SYSTEM	31.6	11.5	31.6	13.5
FLIGHT NEP SYSTEM (ONE)*	43.7	15.9	43.7	18.8
GROUND SUPPORT EQUIPMENT	26.3	9.5	26.3	11.3
LAUNCH AND MISSION OPERATIONS**	1.9	0.7	1.9	0.8
SUB TOTALS	$275.0	100.0%	$233.0	100.0%
FACILITIES	35.0		35.0	
TOTALS	$310.0		$268.0	

- 1972 DOLLARS
- NO ESCALATION
- NO FEE
- NO U-235 COSTS

*INCLUDES ONE SPARE REACTOR AT $8.5M

**TWO SHUTTLE/CENTAUR LAUNCHES AT $13.7M EACH NOT INCLUDED

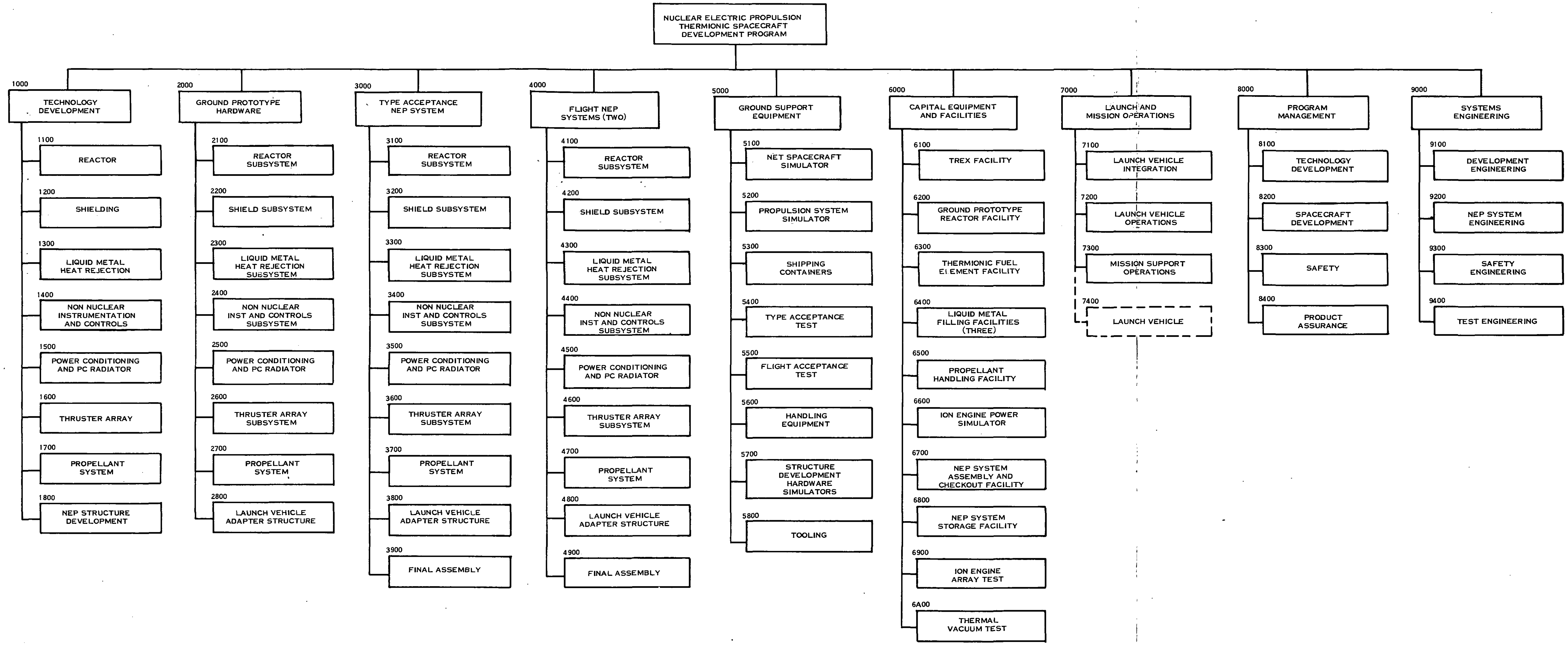


Figure 8-9. Top Level Work Breakdown Structure

The TREX and Ground Prototype Reactors are employed. The GPR also demonstrates the flight primary heat rejection loop, the shield and the flight reactor control system. The secondary heat rejection loop of the power system (i.e., heat pipes) is prototyped independent of the reactor. The thrust system is derived from SEP technology. However, both partial and full ion engine array prototype systems are tested. These employ SEP technology in the NEP configuration and include complete power conditioning and thrust vector control systems.

Type acceptance is performed on a flight-configured NEP system, except that mechanical and electrical reactor simulators are employed. Thermal-vacuum performance is established during an extensive NEP system structural development program.

The minimum NEP system development program objective is to provide a NEP system with a 10,000 hour full power capability for early 1980's missions.

This NEP system could perform most identified interplanetary missions with reduced science payloads or extended mission trip times. It could perform the Comet Halley rendezvous mission, but the risk would be increased. As an objective, this development approach minimizes annual costs through the first five years of the program. The approach employed assumes a high level technology development effort, with emphasis on nuclear component development.

The TREX reactor is not included in this program in order to reduce costs early in the program. TFE life capability is demonstrated in the TFE development program, using TRIGA-type test reactors. A Ground Prototype Reactor demonstration is included in the program which does impact the design of the Flight NEP system. The GPR test includes flight-type primary coolant loop, shield, and reactor control system.

The thrust system is derived completely from SEP technology, except for partial array performance tests in the NEP configuration. Power conditioning nuclear environment tests are also included.

8.2.2 COST STRUCTURE

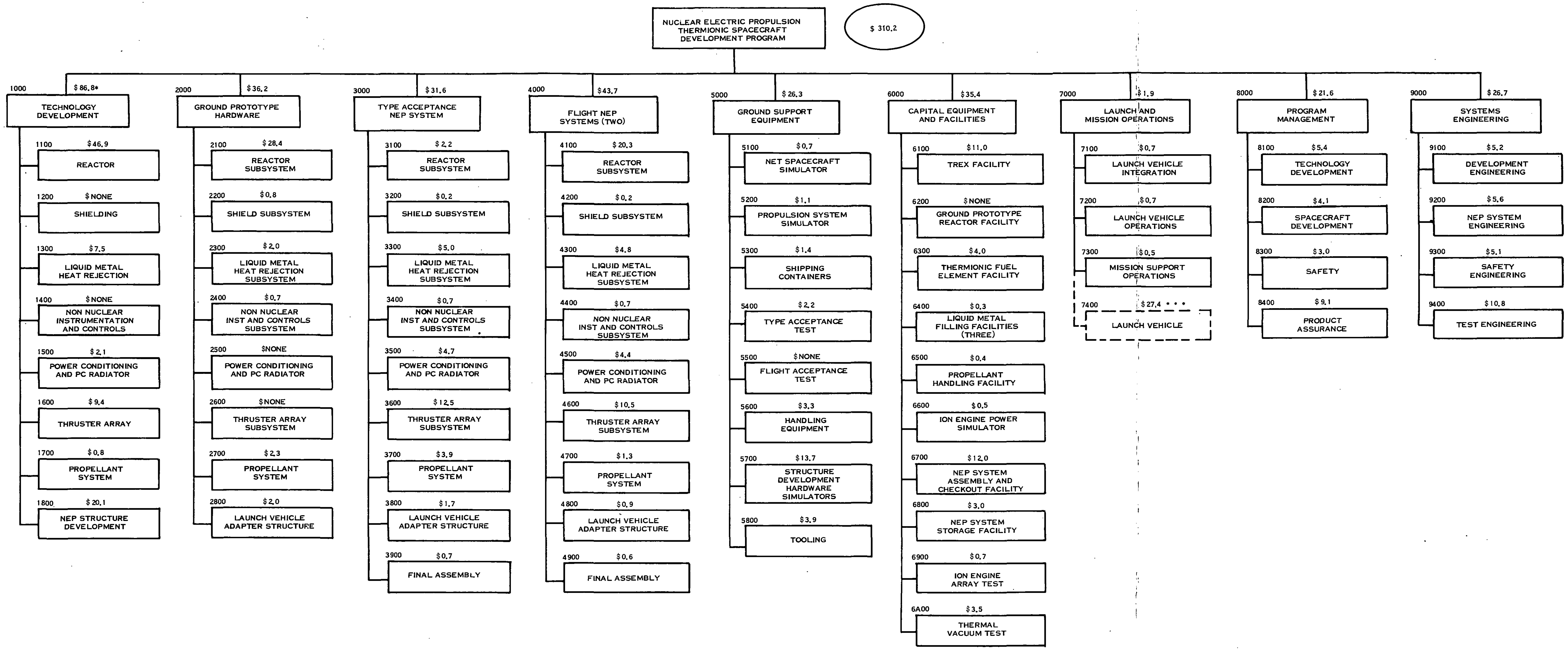
The second level of detail for the cost estimates are presented for the baseline and minimum NEP System development programs in Figure 8-10 and 8-11, respectively. Note that the subtask structure is essentially identical for Task 1000, Technology Development; Task 2000, Ground Prototype Hardware; Task 3000, TA NEP System, and Task 4000, Flight NEP System. Therefore, the total development-through-flight costs of any major NEP subsystem can be readily determined. For example, this total for the Liquid Metal Heat Rejection Subsystem is the total of Subtasks 1300, 2300, 3300, 4300, or Launch vehicle costs estimates, $ 27.40 M for two shuttle/Centaur launches, and the cost of employing two destructive reactor tests in the safety program, $ 11.5M (using scrap TFE's from required TFE production) are shown. These are not included in the program totals.

Ground Support Equipment and Capital Equipment and Facility elements are separately identified. These may be augmented or decreased in scope to meet changing program requirements and the cost impact of such changes on the total program can be readily assessed.

8.3 AVIONICS MODULE COSTS

The preliminary cost estimates for the Avionics Module, shown in Table 8-6, are based on an engineering design and prototype test cycle of thirty months. The subsystem cost estimates include the subsystem engineers, technicians, and drafting support required to tailor the design of previously flight proven components to the specific needs of the NEP system. A factor of 68 percent $12.1 M is added to nonrecurring cost items to account for systems integration/test and program management. The recurring costs are increased by a factor of two to account for production design review during the manufacturing cycle.

The preliminary non recurring cost estimates for the Avionics Module $30. 0 M are while the recurring unit cost estimates total $ 6. 6 M.

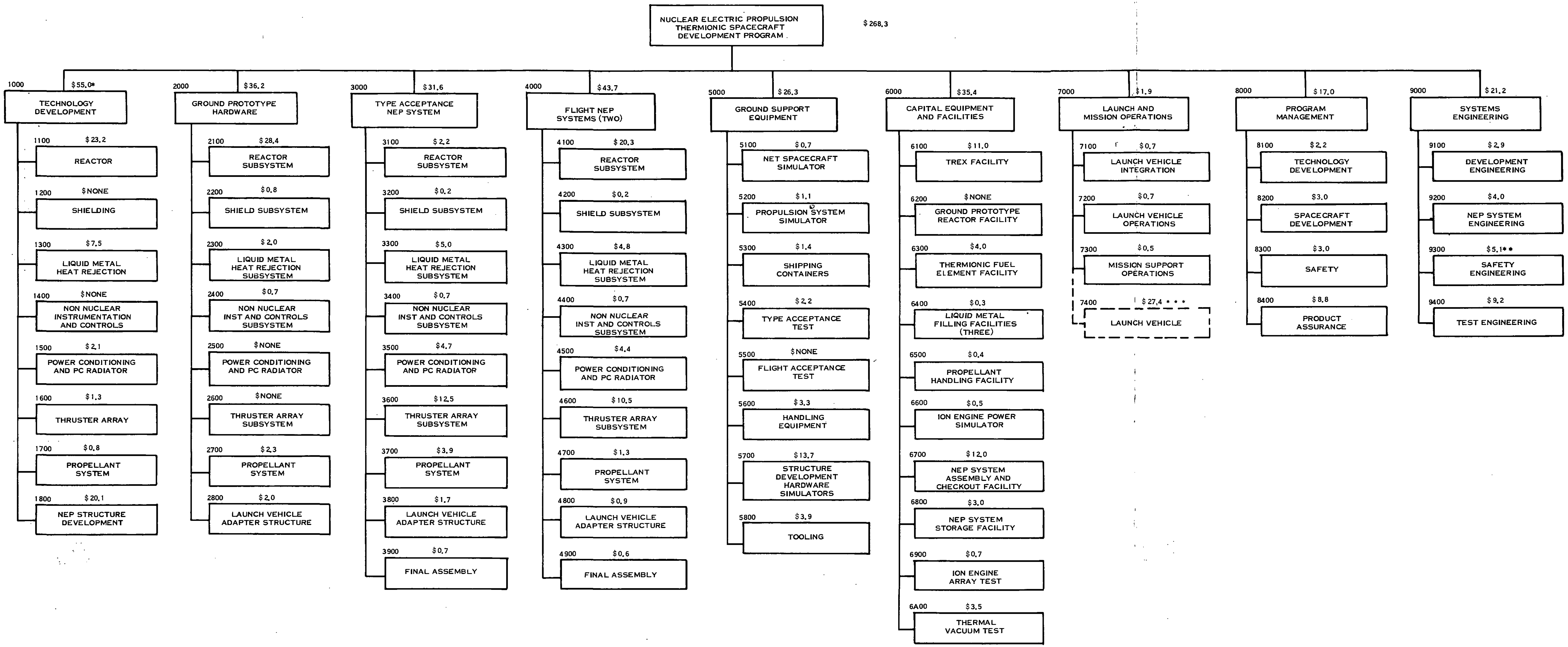


* DOLLARS, MILLIONS

* * ADD $11.5 M IF TWO DESTRUCTIVE REACTOR TESTS ARE INCLUDED (ZERO COST TFE'S)

* * * NOT INCLUDED IN PROGRAM TOTALS

Figure 8-10. Top Level Work Breakdown Structure Technology - Systems and Prototype



* DOLLARS, MILLIONS

* * ADD $11.5 M IF TWO DESTRUCTIVE REACTOR TESTS ARE INCLUDED (ZERO COST TFE'S)

* * * NOT INCLUDED IN PROGRAM TOTALS

Figure 8-11. Top Level Work Breakdown Structure Technology and Systems Analysis Minimum Program

Table 8-6. Avionics Module Cost Estimates

Subsystem	Nonrecurring Costs (Development and Qual.)	Recurring Costs
Attitude Control	$ 5.0 M	$ 1.2 M
Auxiliary Propulsion	2.5	0.5
Communication	4.5	0.4
Video/Lighting Platform	1.5	0.2
Scanning Laser Radar (SLR) Vehicle	3.0	0.3
Structure	0.4	0.2
Thermal Control	0.4	0.2
Mechanisms	0.4	0.2
Power Distribution	0.2	0.1
Subtotal	17.9 M	3.3 M
Product Revision, System Integration & Test, and Program Management	$ 12.1 M	$ 3.3 M
Total	$ 30.0 M	$ 6.6 M

8.4 ADDITIONAL COST CONSIDERATIONS

Many other factors impact the overall NEP system costs. The cost of extensive use of beryllium structure and recurring costs associated with the liquid metal heat rejection subsystem as well as for the total NEP system were investigated and are presented in the following sections.

8.4.1 BERYLLIUM STRUCTURE

The impact of extensive beryllium structure on the NEP system development costs, and upon NEP system units costs was assessed. As noted in Table 8-7, the use of beryllium structure will increase the NEP system development costs by approximately $15 million. The bulk of these costs are associated with the NEP thrust and propellant systems structural development and related structural simulators; the dynamic, mass, and engineering (fit) mock-up. If a beryllium-stainless radiator is required only $1.0-million is related to the development of the power system.

The extensive use of beryllium in production-type NEP systems such as the CNS Test, TA hardware and flight NEP systems will increase unit NEP system costs by about $3.75-million. More than 25 percent ($1.0 million) is tied up in the required NEP System-to-Centaur adapter structure. The largest contributor is the Thrust System ($ 2.10 million). Unit beryllium stainless main heat rejection radiators add about $400 thousand. At the 120 kWe electric power level evaluated, the specific unit cost for beryllium structure is about $30K/kWe.

The identified impact of extensive use of beryllium structure on NEP system development and unit costs is independent of the extent of other development imposed; whether a minimum or baseline program.

Table 8-7. Beryllium Structure Cost Assessment

NEP System Development	
Beryllium Stainless Radiator	$ 1.25 M
NEP System Structure	10.00
Structural Simulators	4.00
Total	$ 15.25 M

NEP System Unit Costs	
Beryllium Stainless Radiator	$0.40 M
Beryllium Ion Engine Array Structure	0.75
Power Conditioning Radiator and Support Structure	1.35
NEP System - Centaur Adapter Structure	1.00
Total	$3.50 M

8.4.2 RECURRING COSTS

To investigate the impact of recurring costs the results of work performed under a separate, but related, contract to take the liquid metal heat rejection subsystem (X3XX) and estimate the cost of producing eight additional units in eight more years after the first two spacecraft were delivered are presented.

The liquid metal heat rejection subsystem was assumed to consist of the primary (reactor) loop ducting and accumulator, four independent radiator loops and their associated ducting, accumulators, and radiator sections, the intermediate heat exchanger (which separate the primary loop from the radiator loops), and a pair of EM pumps in series to drive all five loops. Guidelines and assumptions used in this study are presented.

It was assumed that the flight hardware built and flown as a result of the development program (the first two spacecraft) were acceptable with no additional engineering changes. In one sense this is unrealistic, since no series of spacecraft has been completely frozen as

as to design after only two launches. However, it was a ground rule for this study. The availability of jigs, tooling, and fixtures is, therefore, assumed.

The initial costs were figured on the basis of the requested one per year production rate. GE manufacturing consultants felt that this schedule precluded any real learning-curve gains, and felt that a compressed schedule might lower total cost significantly as long as extra facilities were not required.

As was the case during the development program, no full-power high-temperature testing is employed.

The most serious problem was establishing cost estimates for the production of components and systems which would be the subject of a multimillion dollar development program, and whose design would not be fixed for a minimum of five years. The approach selected involved two separate techniques. First, the production costs estimated in the 6300 series tasks of the development program provided a basis for a per-copy price for a system; however, it was initially thought that these costs might be unrealistic for true production manufacturing. Therefore, a second estimate was obtained in quite a different fashion.

The General Electric Company has a group of Corporate Consulting Services which can be used by Company components to augment their own expertise. These personnel, Manufacturing Engineering Services (MES), independently evaluated the cost of producing the components required for the liquid metal heat rejection subsystem. For each component, a sketch or design of a similar component was selected. These designs were either ones built for testing under NASA contract, or designed as part of a proposal or study effort. For example, the EM pump was based on one designed for a thermionic reactor system proposal while the accumulator was based on a SNAP-8 design.

The results of the liquid metal heat rejection subsystem recurring cost study are summarized in Table 8-8. These costs are markedly lower than those based on the development program, even when the cost of fixtures, tooling, and jigs are accounted for.

Table 8-8. Liquid Heat Rejection Subsystem Unit Cost Comparison

Item	Per Unit Cost	
	Development Program (63XX)	Production Cost
Radiator	$ 1934K	$ 889K
Heat Exchanger	200	93
EM Pumps	800	200
Accumulators	501	115
Ducting	300	76
Total	$ 3735K	$ 1373K

Design and Q/C Costs are not included in this estimate.

The per-copy cost of a complete liquid metal heat rejection subsystem is estimated at $3.7M based on the development program, with a total of about 180,000 man-hours. This value does not include design and Quality/Control costs, which bring the total to $4.035 M.

The comparison of Table 8-8 clearly shows the difference between the unit costs from the development program, and those from the production program.

A detailed comparison of the cost for the main heat rejection radiator (typical of the results for the other components) is shown in Table 8-9.

It is immediately apparent that the manpower levels specified are very different. Three reasons help explain the discrepancy. First, the "First Set" estimates were made with the feeling that reasonable development work would be needed. Second, manufacturing processes still need development and third, these "First Set" units are assumed to be produced in a one-of-a-kind environment.

Economically it is advantageous to compress the schedule up to the point at which additional jigs, fixtures, and major tooling were required. In this way, maximum learning on the part of the workmen would occur. In addition, the test and engineering function would be fully occupied with an additional saving.

In all of the work on this program, the high cost of fabricated beryllium showed up as a major item. If the missions planned can afford to use heavier material such as copper-stainless steel for the radiator fins, approximately $400K/unit can be saved.

The first variation shows a savings of $185K per heat rejection subsystem unit or about 13 percent while the second variation shows a savings of $400K/unit or 29 percent. The results of this portion of the study show that a very real savings can be achieved by freezing the design of the liquid metal heat rejection subsystem and going into a limited production mode.

Estimated total recurring costs for the Flight NEP System are presented in Table 8-10. The first Flight NEP System costed for the development program options totals $35.2M. It is estimated that the cost of the second of these two units is about 80 percent of the cost of the first unit, if these two are built consecutively over a two-year period. It is possible that the cost of subsequent units could approach $25M, or 70 percent of the cost of the first unit.

Table 8-9. Liquid Metal Heat Rejection Subsystem Cost per Unit, Radiator

Man-Hours	First Set Flight Hardware (Less Tooling)	Production Cost (8 in 8 Years)
Engineer	6300	2000
Technician	25060	3460
Shop	57200	13250
Total	88560	18710
Costs ($000)		
Applied Labor	556	108
Overhead	667	130
Labor Cost	1223	238
Material	600	570
Subtotal	1823	808
G&A	182	81
Total Cost	2005	898

Table 8-10. Estimated Recurring Costs
NEP System

First NEP Flight Systems	$ 35.2 M
Second Flight NEP System at ~ 80 Percent	$ 28.2 M
Subsequent Flight NEP Systems May Approach ~70 Percent	$ 24.6 M

SECTION 9

REFERENCES

1-1 "Baseline Tug Definition Document," NASA-MSFC, March 15, 1972.

1-1a "An Interim Report On Comet Rendezvous Opportunities," IIT Research Institute Technical Memorandum #T-21, November, 1969.

2-1 W.Z. Prickett, "A 120 kWe Thermionic Reactor Ion Propulsion Spacecraft for the Comet Halley Rendezvous Mission," 1970 Thermionic Conversion Specialists Conference.

2-2 "Thermionic Spacecraft Design Study - 120 kWe Nuclear Electric Propulsion System," Final Report, General Electric Company, GESP-648, June 30, 1971.

2-3 J.J. Volkoff, "Protection Requirements for the Resistance of Meteoroid Penetration Damage of Interplanetary Spacecraft Systems," Technical Report No. 32-410, Jet Propulsion Laboratory, July 1, 1964.

2-4 E.V. Pawlik, R. Goldstein, D.J. Fitzgerald, "Ion Thruster Performance Calibration," AIAA Paper 72-475, 9th Electrical Propulsion Conference, Bethesda, Maryland, April 17-19, 1972.

2-5 Personnel communication with J.W. Stearns, Contract Technical Monitor at JPL, October 24, 1972.

3-1 Letter Communication, F.K. Mynatt, ORNL; Shield Analysis by ORNL through AEC-HQ, Germantown, Maryland, September 23, 1969.

3-2 E.E. Gerrels, R.E. Killen, "Potassium Rankine Cycle Vapor Chamber (Heat Pipe) Radiator Study," Prepared by the General Electric Company for NASA Lewis Research Center, NASA CR-1866, September, 1971.

3-3 "Thermionic Spacecraft Design Study," Phase I Report, General Electric Company, GESP-7031, January 20, 1970.

3-4 T. Masek, "Sizing a Solar Electric Thrust Subsystem," NASA TR 32-1504, November 15, 1970.

3-5 "Study of Application of Remote Manipulation to Satellite Maintenance," General Electric Company, NASA Report No. CR-73-399, GE Document No. 69SD4286, June, 1969.

3-6 Personal Communication with Bill Reed and Bert Willis of NASA/MSFC, July 19, 1972.

3-7 Bustemente, Davies, "Data Transmission System for Deep Space Probes," AIAA, 5th Annual Meeting, Philadelphia, Pa., October, 1968.

3-8 "Baseline Tug Definition Document," NASA-MSFC, March 15, 1972.

4-1 "Thermionic Spacecraft Design Study," Phase I Report, General Electric Company, GESP-7031, January 20, 1970.

4-2 "Thermionic Spacecraft Design Study - 120 kWe Nuclear Electric Propulsion System," Final Report, General Electric Company, GESP-648, June 30, 1971.

4-3 Personal Communication, Dr. C.D. Sawyer, JPL, May 1971.

4-4 Personal Communications, J.W. Stearns, JPL, August 1972.

5-1 Personal Communication with F.R. Schmidt at NASA/HQ, November 6, 1972.

5-2 "Baseline Tug Definition Document," NASA-MSFC, March 15, 1972.

A-1 "Nuclear Electric Propulsion Mission Engineering Study, Phase I Review," General Electric Company, GESP-7069, November 5, 1971.

A-2 "Thermionic Spacecraft Design Study, 120 kW Nuclear Electric Propulsion System," Final Report, General Electric Company, GESP-648, June 1971.

A-3 "Nuclear Electric Propulsion Mission Engineering Study, Phase III Review," General Electric Company, GESP-7077, August 31, 1972.

A-4 W.W. Phelan, "Impact of Jupiter's Radiation Environment on NEP Spacecraft Mission and Shielding Requirements," PIR-3871, February 10, 1972.

A-5 Neil Divine, "Jupiter Radiation Belt Engineering Model," Proceedings of the National Symposium on Natural and Man-Made Radiation in Space, NASA TM X-2440, January 1972.

A-6 "Thermionic Spacecraft Design Study," Phase I Report, General Electric Company, GESP-7031, January 1970.

APPENDIX A

DESIGN SPECIFICATIONS FOR THE THERMIONIC NUCLEAR ELECTRIC PROPULSION MULTI-MISSION STAGE

A.1 OBJECTIVES

The primary objective of this appendix is the definition of the design objectives and criteria of a 120 kWe (Pe) internal fuel (flashlight) thermionic reactor unmanned electric propulsion system with multi-mission capability. The baseline system shall deliver 120 kWe power to the thrust system, and shall be capable of being installed on the Centaur Stage within the standard Shuttle cargo bay. Minimum weight for the 120 kWe electric propulsion system is a design objective.

A.2 BASELINE MISSIONS

The propellant capacity and life characteristics of the multi-mission NEP Stage will be designed to perform a family of outer planet exploration and comet rendezvous missions. The selected baseline interplanetary missions are the Comet Halley rendezvous mission and the tight Jupiter orbiter (terminal circular orbit at 5.9 R_J). The transportation of operational payloads to and from synchronous equatorial earth orbit is the baseline mission for geocentric orbit applications.

A.3 GENERAL DESIGN GUIDELINES

The baseline NEP Stage (depicted in Figure A-1) is an end thrust configuration having a thrust vector parallel to the vehicle's major axis. The selected arrangement places the reactor at the extreme aft end of the vehicle with maximum separation from the power conditioning, guidance/control and communication equipment, and the payload.

Electrical power is generated by a 22 to 24 volt, internally fueled, thermionic reactor with the waste heat dissipated by a heat rejection system consisting of a pumped primary loop and a heat pipe radiator. A thruster array, composed of approximately twenty-four 30 cm diameter mercury electron bombardment ion engines (including four spares), converts approximately 110 kilowatts of electrical power to approximately 85 kilowatts of beam power for propulsive thrust.

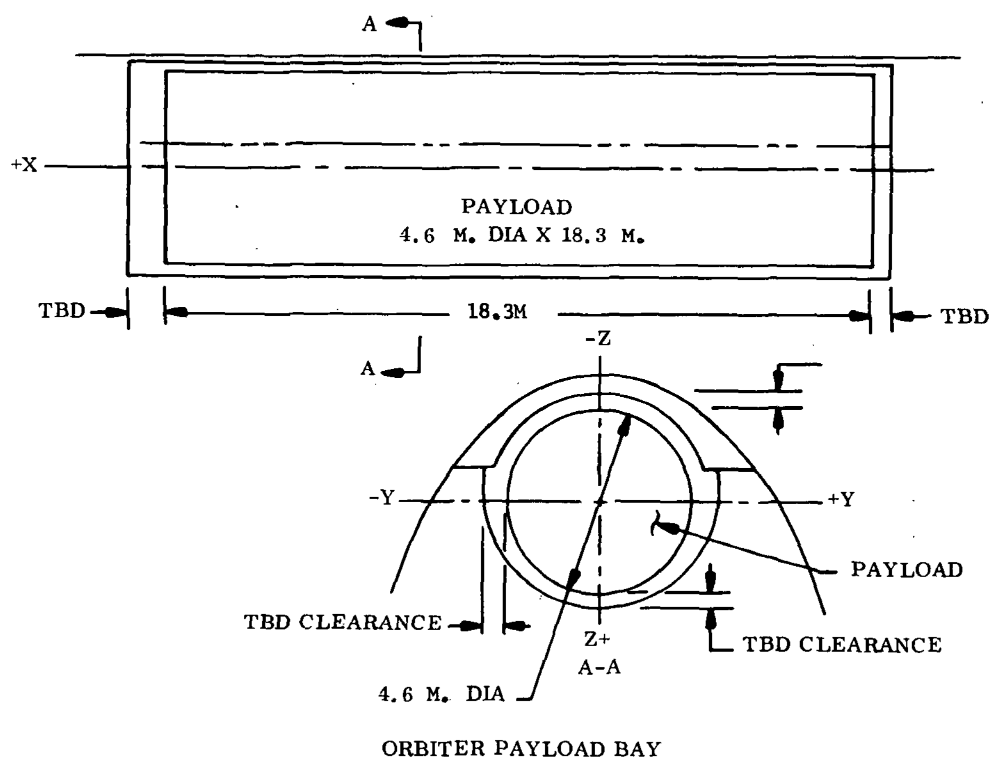


Figure A-1. Multi-Mission NEP Thermionic Stage (120 kWe)

The following design guidelines have been established as a reference for the design definition of the multi-mission thermionic electric propulsion stage (References A-1, A-2, and A-3):

1. Net power of 120 kWe shall be supplied to the thrust subsystem.

2. Thrust is provided by an array of 30 cm mercury ion engines, including 20 percent spares, each of which delivers 4000 seconds specific impulse. JPL TM 32-1504 is used as the basis for the thruster design.

3. The mercury propellant requirements are a function of the mission. The mercury propellant tanks will be sized to hold the complete propellant inventory requirement, plus a 10 percent margin.* The required propellant inventory (not including the 10 percent margin) is given by the relation,

$$M_p = \frac{(134/\eta pc)P_e t}{315 + V_e 2}$$

V_e = $g\ I_{sp}$, km/sec

M_p = Mercury required, kg

P_e = Power to Thrust System, kWe

t = Thrust Time, Days (full power)

η_{pc} = Power Conditioning System Efficiency

4. Beryllium or magnesium panels will be used for the power conditioning radiator panels.

5. As an objective, the cost/mass of the avionics module will be minimized. Consideration will be given to low cost design techniques.

6. The thermionic reactor currently being developed by GGA for the AEC is the reference reactor.

7. The heat rejection subsystem is to be a pumped primary loop (NaK filled) with sodium heat pipes forming the secondary heat rejection system.

*The requirement to include the 10 percent propellant margin is optional and will reduce the maximum payload capability of the NEP Stage.

8. The heat rejection subsystem shall be designed to have a 99 percent confidence of meeting the design specifications at the end of 50,000 hours. Wherever possible, credit shall be taken for meteoroid protection afforded by materials which can act as bumpers. Armor requirements are based on the earth orbital meteoroid environment. The meteoroid protection requirement will be compatible with the following models:

I. Penetration Model for Armor

$$t = 0.5m^{0.352}\rho_m^{1/6}v^{0.875}$$

where,

t = armor thickness, cm

ρ_m = meteoroid density, gm/cm^3

m = meteoroid mass, gm

v = meteoroid velocity, km/sec

II. Meteoroid Flux*

$$\varphi = am^{-\beta}$$

where,

φ = cumulative meteoroid flux, number of particles/m^2-sec.

a = empirical coefficient $\qquad$ β = empirical exponent

m = meteoroid mass, gm

The baseline data listed below is used in conjunction with the previous models to calculate an equivalent near-earth meteoroid protection requirement:

$\rho_m = 0.5\ gm/cm^3$ $\qquad$ $a = 6.62 \times 10^{-15}$

$v = 20$ km/sec $\qquad$ $\beta = 1.34$

*Effective February 1, 1973, this meteoroid flux model will be replaced by the MJS Meteoroid flux model.

The LiH neutron shield, the mercury propellant system, the heat rejection loop and the individual power conditioning modules shall be protected from meteoroid damage. The radiator models to be used will be developed from the SPARTAN Series computer code results and will be based on the preceding earth meteoroid protection requirements.

9. Maximum allowable solid state electronic component and radiator temperature is $373^{o}K$ ($212^{o}F$).

10. Power conditioning radiator is sized for a mean near earth heat sink temperature of $253^{o}K$ ($-5^{o}F$).

11. Maximum allowable ion engine temperature is $523^{o}K$ ($480^{o}F$).

12. Maximum allowable neutron shield temperature is $755^{o}K$ ($900^{o}F$). As an obejctive, the minimum allowable LiH neutron shield temperature during operation is $644^{o}K$ ($700^{o}F$).

13. Maximum allowable EM pump winding temperature is $644^{o}K$ ($700^{o}F$).

14. Reactor controls power requirement is 0.8 kWe.

15. At least 1 kWe is allocated for operation of the avionics subsystem.

16. Cesium reservoir temperature control power requirement is 0.5 kWe.

17. All pumped liquid metal coolants are NaK-78.

18. Individual power conditioning modules are to be provided for each ion engine.

19. Payload, power conditioning and communications will be shielded to an integrated dose not to exceed 10^{12} nvt (>1 MeV) and 10^{6} rads gamma. Credit will be taken for attenuation by non-shielding materials.

20. As an objective, no permanent gamma shielding will be employed in the baseline stage design.

21. For the Jupiter orbiter mission, the Jovian trapped radiation environment will be based on a onminal electron and proton flux model (References A-4 and A-5) since, pending more substantial data expected from Pioneer F and G, this is considered to be more representative than an upper limiting model.

22. The NEP Stage will be designed to be compatible with the 18.3 m long by 4.6 m diameter Space Shuttle cargo bay dimensions when installed on the 9.1 m long Centaur launch stage. This may result in the design of a "deployable" NEP Stage configuration.

A.4 MULTI-MISSION NEP STAGE REQUIREMENTS

The NEP Stage consists of a propulsion system and an avionics subsystem. The subsystems that comprise the propulsion system are the power subsystem, thrust subsystem, and propellant subsystem. The key subsystems that comprise the avionics subsystem are the attitude control subsystem, flight command subsystem, flight telemetry subsystem, video/lighting subsystem, docking subsystem, and the thermal control subsystem.

The thermionic reactor must supply the hotel loads, such as liquid metal pumping and powerplant control, and must provide the required 120 kWe to the thrust subsystem. The design of the NEP Stage must provide 20 percent redundance in the electric power capability of the reactor, ion engines, and related power conditioning.

A.4.1 POWER SUBSYSTEM

The components that comprise the power subsystem for the ~ 23 volt multi-mission NEP Stage are:

1. Reactor
2. Heat Rejection Subsystem
3. Reactor Radiation Shield
4. Electrical Subsystem
5. Structure

Brief description and system requirements are presented in the following paragraphs.

A.4.1.1 Reactor

The physical characteristics of the ~23 volt Flashlight Reactor will be supplied by the NASA/AEC Joint Office.

A.4.1.2 <u>Heat Rejection Subsystem</u>

The heat rejection subsystem is to consist of a pumped primary loop with a heat pipe radiator (i.e., secondary loop). The subsystem components are:

1. Sodium heat pipe radiator
2. AC induction EM pumps (two)
3. Accumulator(s)
4. Piping
5. NaK coolant (primary loop)

Since the reactor has been designed to accommodate 20 percent diode losses, the primary radiator is to be capable of operating at the more severe end-of-mission thermal load. The accumulator(s) will provide for primary loop coolant expansion and pressurization.

A.4.1.3 <u>Reactor Radiation Shield</u>

In accordance with the established guidelines, the power conditioning and communications electronics (and the science payload for interplanetary mission) will be shielded to integrated dose limits of 10^{12}nvt (E_n > 1 MeV) and 10^6 rads gamma. The integrated radiation dose will consist of that from the reactor plus the contribution attributed to the space environment (Van Allen and/or Jovian radiation).

A.4.1.3.1 Neutron Shield

The neutron shield will consist of a lithium hydride stainless steel honeycomb enclosed in a stainless steel can. The lithium hydride will perform most of the required neutron shielding with additional neutron attenuation contributed by the liquid mercury propellant. If auxiliary cooling of the shield is required to maintain the shield temperature below 755°K (900°F) (Reference A-6), these requirements are to be minimized by the use of heat pipes. To prevent swelling from neutron damage, the LiH shield temperature should be maintained above 644°K (700°F) during operation.

A.4.1.3.2 Gamma Shield

The primary gamma shielding for the 22 to 24 volt Flashlight Reactor is provided by the liquid mercury propellant. The propellant tank mean diameter is to be such that the initial propellant thickness maximizes the gamma shielding requirements. Therefore, the need for permanent, heavy gamma shielding, such as tungsten or depleted uranium, can be minimized. If auxiliary cooling of the stored liquid mercury is required, a heat pipe system shall be employed to reject heat.

A.4.1.4 Electrical Subsystem

A segmented transmission line of copper cable, aluminum bus bar, and aluminum cable carries the ~ 23 volt electrical power from the reactor to the main power conditioning modules. Copper cables connect the reactor to aluminum bus bars which transmit the electrical power from the forward face of the shield to the PC radiator. At that point, aluminum cables carry the power to the PC modules.

The temperature extremes of the low voltage cable are $900^{O}K$ ($1160^{O}F$) and $373^{O}K$ ($212^{O}F$). The low voltage cables will be designed such that no additional heat load will be placed on the electronics as a result of that generation in and/or conduction down the cables.

The electrical subsystem also includes the hotel power conditioning equipment for the EM pumps along with associated cooling radiator and power cabling to the pumps and reactor control actuators. The hotel power conditioning is to be based on previously investigated components (Reference A-6). It will supply variable frequency AC power to the EM pump(s) at a conversion efficiency of 90 percent.

A.4.1.5 Support Structure

Power subsystem structural elements are required in two general areas:

1. Support and attachment members for the reactor, radiation shield and heat rejection components.

2. Strengthening rings, etc., for the primary radiator.

Circular frames and supporting clips and attachments are needed to maintain primary radiator structural integrity under the expected launch load, imposed by the Shuttle/Centaur launch system.

Additional structure may be required for a deployable stage design to assure compatibility with the Shuttle cargo bay dimensionsal limitations.

A.4.2 THRUST SUBSYSTEM

The thrust subsystem consists of the following major components:

1. Power conditioning modules
2. Ion engines
3. Power conditioning radiator
4. PC to ion engine high voltage transmission cables
5. Structure

A.4.2.1 Main Power Conditioning

Power is delivered from the reactor leads at a potential of ~23 volts and is distributed to the power converters. The 27 converters (one for each of the 6 TFE units) change the low voltage DC output of the thermionic reactor to squarewave AC, and transform the ~23 volt reactor output to ~2000 volts for use by the main power conditioner for the ion engines. With individual power conditioners for each thruster, compensation for engine arcing is provided within the control circuit of each conditioner. Some of the ~23 volt input to the inverters is transformed to ~50 volt for input to the auxiliary hotel power conditioners.

The function of the power conditioning radiator is to maintain desired operating temperatures of $373^{o}K$ ($212^{o}F$) in the power conditioning modules by dissipating the heat generated in the modules via direct radiation to space.

The power conditioning radiator is to consist of magnesium or beryllium panels, joined to form a multi-sided, right-angle prism. The PC modules are to be distributed on the inner surface of the radiator in axial bays.

A.4.2.2 Ion Engines

The thruster array is to consist of approximately twenty-four 30 cm mercury ion engines, the Thruster Vector Control (TVC) system, and their immediate support structure. The definition of the number, size, and arrangement of the electron bombardment mercury ion engines must consider the following guidelines and constraints:

1. 120 kWe (P_e) is delivered to the main power conditioning for distribution to all operating ion engines.

2. The number of ion engines must include 20 percent redundancy.

3. JPL TM 32-1504 will be the basis for the thruster design.

4. Adequate thermal control must be provided for the ion engines.

5. Approximately 50 percent of the ion engines must be gimbaled to provide for roll TVC about the thrust axis. (Pitch and yaw control can be achieved by monitoring the thruster array on hinged panels or by the incorporation of variable thrust ion engines.) The ion engine spacing must permit rotation of the gimbaled ion engines ± 10 degrees. The spacing of gimbaled ion engines requires special consideration in order to accommodate the gimbal mechanism, based on designs being developed.

6. The number and size of the ion engines must be compatible with the utilization of a fixed amount of propellant over a fixed thrust time.

A.4.2.3 High Voltage Transmission Cables

The high voltage aluminum cables transmit the electrical power from the PC modules to the ion engines.

A.4.2.4 Support Structure

Thrust system structural components are required in three general areas:

1. Support and attachment members for the ion engine thruster array.

2. Support and attachment members for the power conditioning modules and power conditioning radiator.

3. Docking assembly.

Additional structure may be required for a deployable stage design to assure compatibility with the Shuttle cargo bay dimensions limitations.

A.4.3 PROPELLANT SUBSYSTEM

The propellant subsystem consists of the mercury propellant, its containment tanks, and the propellant distribution system. The mercury propellant is located in a stainless steel tank forward of the LiH neutron shield.

The tank design is to provide for positive mercury expulsion via a metal bellows system pressurized by a cold gas system. This will assure that no voids will form in the tank, which, if incurred, would result in radiation streaming. The propellant tank volume shall be capable of containing 110 per cent of the required propellant mass. This additional mass may not be loaded at launch.

A.4.4 AVIONICS SUBSYSTEM

The avionics subsystem serves as the command and control module of the NEP Stage during in-flight operation. As an objective, the requirement for commonality between interplanetary and geocentric missions is expected to lead to the selection and development of one (or at the most two) avionics subsystem for all missions. The key subsystems contained in this module include attitude control, flight command, flight telemetry, video/lighting, docking, and thermal control. A mass of up to 600 kg and a power level of up to 1 kWe is allocated for the avionics subsystem.

The total operational lifetime of the avionics subsystem is to be 50,000 hours.

A.4.4.1 Attitude Control Subsystem

The Attitude Control Subsystem (ACS) is to consist of the Thrust Vector Control (TVC) Subsystem, the Reaction Control Subsystem (RCS), and the sensors and trackers required to

provide the vehicle attitude and thrust vector orientation functions necessary to satisfy mission requirements. The ACS design requirements must be capable of being accommodated over a wide range of vehicle mass property values, dependent upon the particular payload which is being transported by the NEP Stage.

A.4.4.2 Flight Command Subsystem

The Flight Command Subsystem (FCS) is comprised of the Central Computer and Sequencer (CC&S) and the Flight Data Subsystem (FDS). The primary function of the central computer and sequencer is to maintain control of the NEP Stage, both thrust vector control and reactor control. The primary function of the FDS is to monitor the operational status of the NEP Stage.

A.4.4.3 Flight Telemetry Subsystem

The Flight Telemetry Subsystem (FTS) is to contain the communication equipment required to provide the vehicle attitude and thrust vector orientation functions necessary to satisfy mission requirements. The ACS design requirements must be capable of being accommodated over a wide range of vehicle mass property values, dependent upon the particular payload which is being transported by the NEP Stage.

A.4.4.2 Flight Command Subsystem

The Flight Command Subsystem (FCS) is comprised of the Central Computer and Sequencer (CC&S) and the Flight Data Subsystem (FDS). The primary function of the central computer and sequencer is to maintain control of the NEP Stage, both thrust vector control and reactor control. The primary function of the FDS is to monitor the operational status of the NEP Stage.

A.4.4.3 Flight Telemetry Subsystem

The Flight Telemetry Subsystem (FTS) is to contain the communication equipment required for the performance of the NEP mission. The RTS is to provide the RF link for four different functions (partially interrelated): telemetry, tracking, command, and rendezvous and docking.

A.4.4.4 Video/Lighting Subsystem

The video/lighting subsystem is to consist of the necessary cameras and lighting equipment to assist in remote rendezvous and docking functions. The requirements for the video picture are to be 250 to 500 lines per inch, greater than ten frames per second, and a signal-to-noise ratio of 40 to 50 dB. The illumination range requirement is to be from full sun to complete shadow.

A.4.4.5 Docking Subsystem

The docking subsystem is to be active for attachment to passive payloads, but will include the mechanism required to convert it to a passive device if safety or other mission considerations should require docking with an active space vehicle which would be assuming primary control. The docking subsystem is to meet the design requirements as defined below:

1.	Miss Distance	$\pm$ 0.15m
2.	Miss Angle (Each Axis)	$\pm$ 1 degree
3.	Longitudinal Velocity Control	0.03 to 0.3 m/sec
4.	Lateral Valocity Control	0 to 0.03 m/sec
5.	Angular Velocity	$\pm$ 0.1 deg/sec

A.4.4.6 Thermal Control Subsystem

The range of the thermal dissipation requirement of the avionics subsystem is wide - from potentially long periods with limited available power, and consequently minimal dissipation, during reactor shutdown, to the condition of maximum dissipation of around 500 watts with unlimited power available during reactor operation. The thermal control subsystem must maintain allowable component temperatures throughout the mission.

A.4.5 NEP STAGE INTERFACES

A.4.5.1 Centaur D-1T

The NEP Stage is to provide an adequate docking support structure to anchor the base of the stage to the Centaur D-1T.

A.4.5.2 Space Shuttle

The NEP Stage is to be designed so that the stage mounted on the Centaur D-1T stage can be placed in the Space Shuttle cargo bay (4.6 m in diameter by 18.3 m long). To fit both of these vehicles in the Shuttle cargo bay, the NEP Stage may have to be deployable. Deployable of the stage can be accomplished after Earth escape velocity is reached.

Any packaging and intergration items that may be required (such as auxiliary electrical power to prevent a NaK freeze-up and an inert gas "blanket" to preclude the possibility of a NaK-oxygen reaction) in the Space Shuttle transport of the NEP Stage are to be incorporated in the "transfer module" that the vehicle is placed on while being transported by the Space Shuttle.

A.4.5.3 Synchronous Orbit Payload

The NEP Stage docking adapter must be capable of mating to synchronous orbit payloads of variable size and mass. These payloads may range in diameter up to 4.6 m and may weigh as much as 4500 kg.

Interfaces between the propulsion system and payload are to be limited to mechanical, electrical, and radiation. These interfaces should be common for all payloads.

A.4.5.4 Interplanetary Science Payload

A mass of 120 kg and a power level of 140 watts is to be allocated for the interplanetary science payload. Typical components included in the interplanetary science payload shall be a meteoroid-asteroid detector, micrometeoroid detector, plasma probe, IR spectrometer, UV spectrometer, plasma wave detector, DC magnetometer, and an imaging TV camera.

The interfaces between the propulsion system and the interplanetary science payload are to be basically the same as those between the propulsion system and the synchronous orbit payload.

GENERAL ELECTRIC

Space Division

Headquarters: Valley Forge, Pennsylvania □ Daytona Beach, Fla. □ Cape Kennedy, Fla. □ Evendale, Ohio □ Huntsville, Ala. □ Bay St. Louis, Miss. □ Houston, Texas □ Newport Beach, Calif.